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FINAL REPORT

SPACE SHUTTLE HYPERGOLIC BIPROPELLANT RCS ENGINE DESIGN STUDY

BELL MODEL 8701

REPORT NO. 8701-910041

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

LYNDON B. JOHNSON SPACE CENTER
HOUSTON, TEXAS

REFERENCE: DATA ITEM T-853-5

MAY 1974

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APPROVED BY: M.L. CHAZEN
PROGRAM MANAGER/TECHNICAL DIRECTOR
RCS ENGINE PROGRAM

NASA TECHNICAL MONITOR: MR. NORMAN CHAFFEE

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FINAL REPORT

SPACE SHUTTLE HYPERGOLIC BIPROPELLANT RCS

ENGINE DESIGN STUDY

T-853-5

1.0 INTRODUCTION

This document comprises the final report of the Space Shuttle Bipropellant RCS Engine Technology Program conducted by Bell Aerospace Company under Contract NAS 9-12996. It is submitted in compliance with Data Item T-853-5.

1.1 Program Objectives

The overall objective of this program was to firmly define the level of the current technology base in the area of N_2O_4/MMH RCS engines suitable for Space Shuttle application. This was accomplished by a program of engine analyses, design, fabrication, and test. The program culminated in this comprehensive final report and in the delivery of test evaluation hardware to NASA. Specific objectives of this program included:

1. Demonstration of the capability to extrapolate current engine design experience to the design of an RCS engine of representative shuttle size having the required safety, reliability, performance, wide off-limits operational capability, and minimum servicing and maintenance requirements.

2. Demonstration of multiple reuse capability.

3. Identification of current design and technology deficiencies and critical areas for future effort.

4. Provide engine design and performance information to guide decisions in the mainstream vehicle program.

5. Provide engine hardware for NASA evaluation.

1.2 Program Scope

The program consisted of five phases as follows:

Phase I	Analysis, supporting tests, and design.
Phase II	Hardware fabrication.
Phase III	Engine testing.
Phase IV	Post-test analysis and design.
Phase V	Test evaluation hardware submitted to NASA.

The baseline engine design for the program was as follows:

Thrust	- 600 lbf
Area Ratio	- 40:1
Chamber Pressure	- 200 psia
Propellants	- N_2O_4/MMH
Mixture Ratio (O/F)	- 1.6

Design requirements are presented in Table I-1.

The engine design is predicated upon the need for long life with minimum servicing. It specifically includes long life, compatible materials; injector/valve of the simplest type, free from contamination traps and having a high degree of visual inspectability; a chamber having wide thermal margins to provide an engine insensitive to feed system anomalies. Specific impulse (steady state and pulse mode) is as high as possible commensurate with the reusability and servicing requirements and thermal margin.

The Bell flight type engine (see Figure 1-1) consists of a film-cooled, insulated, coated columbium alloy thrust chamber and nozzle extension, a columbium injector welded to the thrust chamber, and a direct acting torque motor bipropellant valve. Thermal protection of the engine valve is provided by a titanium alloy standoff which is welded directly to the columbium injector. The injector is of a doublet configuration with vortex fuel barrier and is designed so that oxidizer impingement on the thrust chamber wall is avoided. The design provides a large thermal margin and high reliability. External insulation is accomplished by a low density (12 lb/cu ft) Dynaflex blanket encased in foil and mechanically attached to the engine.

1.3 Performance Goals

The RCS engine technology for Space Shuttle application relates primarily to the ability to translate current design experience over a thrust level range and to the ability

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TABLE 1-1

DESIGN REQUIREMENTS

VACUUM STEADY STATE THRUST (F∞)	600 LBS
EXHAUST NOZZLE AREA RATIO (ε)	40
STEADY STATE CHAMBER PRESSURE (Pc)	200 PSIA
ENGINE STEADY STATE MIXTURE RATIO, (O/F)	1.6
PROPELLANTS	
OXIDIZER	N_2O_4 (MIL-P-26539C MON-1)
FUEL	MMH (MIL-P-27404A)
PRESSURANT	HELIUM
PROPELLANT FEED CONDITIONS	
STATIC PRESSURE	300 ± 6 PSIA
DYNAMIC PRESSURE	290 ± 10 PSIA
TEMPERATURE	75 ± 35°F
VALVE VOLTAGE	28 ± 4 VDC
MINIMUM IMPULSE BIT (NOMINAL CONDITIONS)	30 ± 10 LB-SEC
MAXIMUM PULSE FREQUENCY	5 CPS
MAXIMUM SINGLE FIRING	600 SEC
MAXIMUM FIRING TIME PER MISSION	1000 SEC
MAXIMUM NUMBER PULSES PER MISSION	2000
ENGINE LIFE	100 MISSIONS 10 YEARS 100,000 SEC 200,000 PULSES
STABILITY	
HIGH FREQUENCY	DYNAMICALLY STABLE —RECOVERY IN 20 MS
LOW FREQUENCY	OSCILLATIONS < ±5%
THRUST VECTOR ALIGNMENT	0.5°
MAXIMUM OUTER WALL TEMPERATURE	800°F
ENGINE ENVIRONMENT	
TEMPERATURE	-20°F TO +300°F
LAUNCH/REENTRY TEMPERATURE (5 MINUTES)	2000°F AT NOZZLE EXIT 1500°F AT THROAT
PRESSURE	S/L TO 10-13 TORR.
ACCELERATION (TWICE/MISSION)	3.5g FOR ONE MIN.
VIBRATION	SINUSOIDAL VIBRATION: (6 MINUTES) 5 - 23 HZ 1g 23 - 40 HZ 0.036 IN. D.A. RANDOM VIBRATION: (1 MINUTE/AXIS/MISSION) 20 - 90 HZ $0.1g^2$/HZ 90 - 180 HZ +12DB/OCTAVE 180 - 350 HZ 1.6 g^2/HZ 350 - 2000 HZ -6DB/OCTAVE
SHOCK	TBD
RAIN (PER MISSION)	0.5 IN/HR FOR 0.5 HR
SAND (PER MISSION)	140 MESH-500 FPM (4 HR)
SALT ATMOSPHERE (PER MISSION)	COASTAL AREAS AT 75±20°F FOR 30 DIA.
HUMIDITY (PER MISSION)	0-100% RH FOR 30 DIA.
LEAKAGE OF PROPELLANTS OR COMBUSTION GASES	NONE

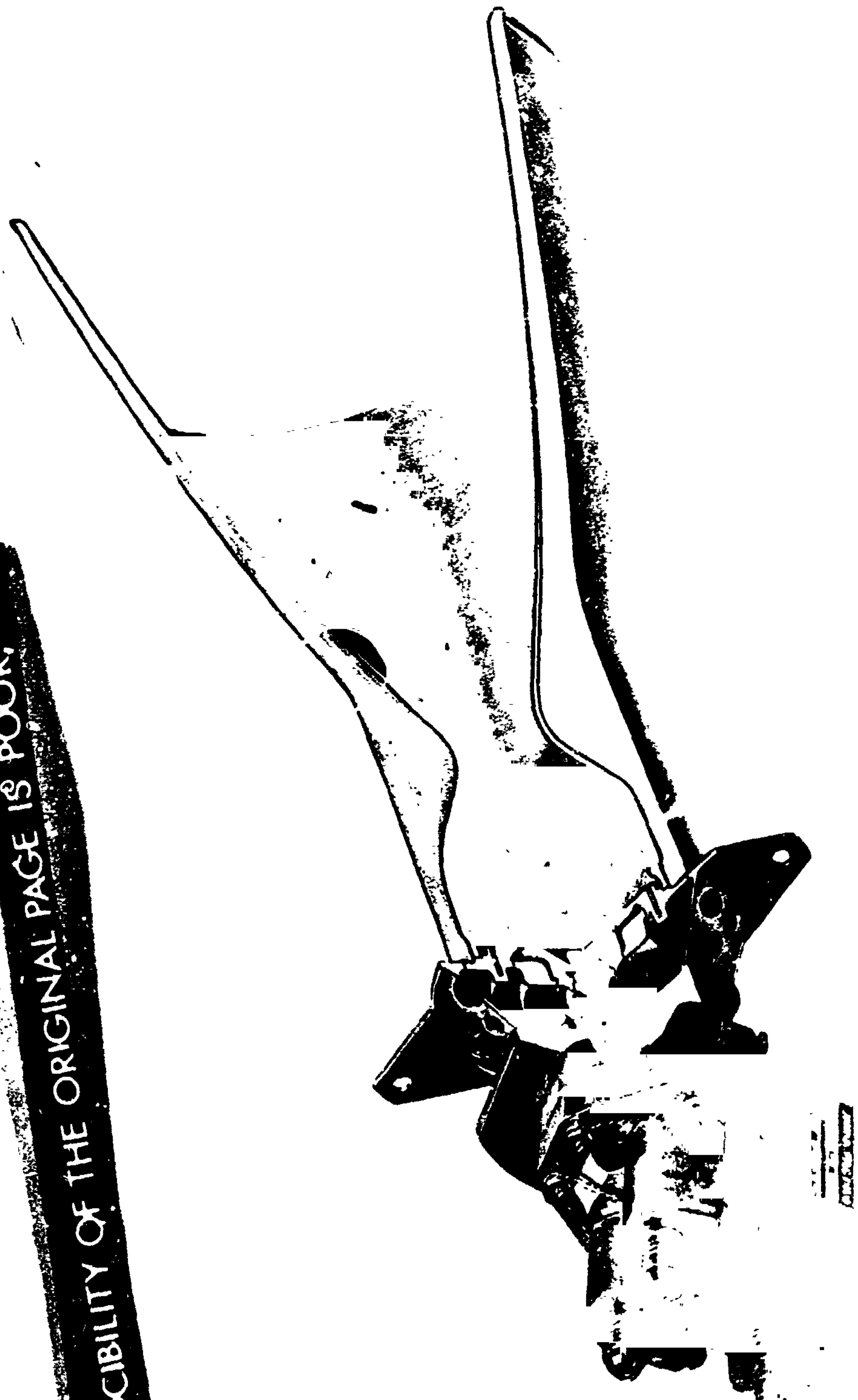

Figure 1-1 600-lb Thrust Engine

of the engine to survive multiple missions over an extended period with minimum servicing and maintenance requirements. The performance goals of the engine can be summarized as follows:

Steady-state specific impulse - 295 pounds force-sec. per pound mass for durations in excess of 1 sec.

Pulse specific impulse - 220 pounds force-sec per pound mass for a 30 pound-sec impulse bit.

Specific impulse shift - minimized over the nominal ranges of propellant feed temperature, pressure, valve voltage, and pulsing frequency.

Mixture ratio shift - minimized from the nominal value of 1.6 over the nominal range of propellant feed temperatures, pressure, valve voltage, pulse width and pulse frequency.

Engine start transient - 50 ms maximum to 90% of rated thrust under nominal operating conditions.

Engine shutdown transient - 50 ms maximum to 10% of rated thrust under normal operating conditions.

Off limits operating capability - capability of broad off limits operation without sustaining damage including feed pressure ranges, feed temperature ranges, valve voltage ranges, valve opening and closing mis-match, operation with gas bubbles, and blockage of injector orifices.

1.4 Program Schedule

The program schedule is presented in Figure 1-2.

SPACE SHUTTLE RCS ENGINE PROGRAM

BASIS: CONTRACT MODIFICATION 4C/5S

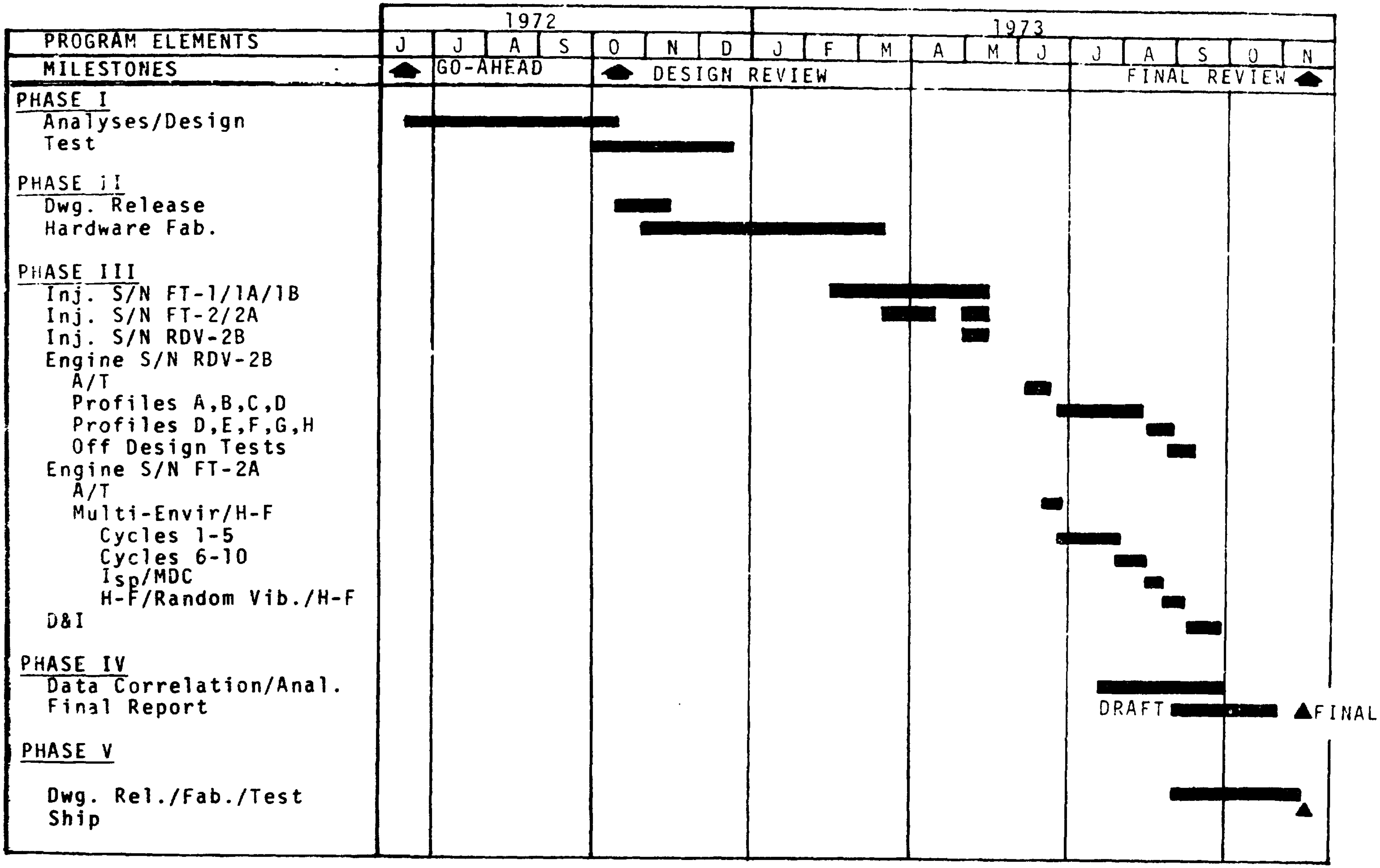

Figure 1-2

2.0 SUMMARY OF RESULTS

In the course of the 17-month program, 17,740 seconds of firing time was accumulated in 18,528 firings on seven Columbium alloy engines. The demonstration test program met or exceeded all performance and durability goals.

One engine assembly (S/N RDV-2) successfully accumulated 10,411 seconds of burn time (10 missions)* in 6377 firings including a 600 second continuous firing and 567 thermal cycles without any engine maintenance. This engine also successfully demonstrated a series of off design tests to evaluate:

a. Helium bubble ingestion

b. Low chamber pressure

c. Off-mixture ratio

d. Nozzle-up attitude

All tests were successful.

A second engine (S/N FT-2) successfully completed 10 simulated mission environmental/hot fire cycles consisting of (1) salt spray, (2) sand and dust, (3) sinusoidal vibration, (4) humidity, and (5) pulse mode and steady-state fire tests.

The testing of both engines was performed without any engine cleaning or maintenance. Thus, the extensive reusability of high performing, low wall temperature, columbium rocket engines without flushing was completely demonstrated. This was probably the most significant result of the technology program.

Detail Results

A comparison of test results with program goals is presented in Table 2-1. Major program results are summarized below:

Engine Reuse, Maintenance and Inspection

Both in Phase I and Phase III engines were subjected to repeated mission duty cycle firings (4 in Phase I and 10 in Phase III) without engine maintenance with no adverse

*Equivalent to 52 mission cycles as defined by Space Division of Rockwell International - (RI/SD).

effects. One Phase III engine was also subjected to repetitive environmental testing which consisted of 10 series of exposures to (1) salt spray, (2) sand and dust, (3) sinusoidal vibration, (4) humidity and (5) hot firings. Post-test sectioning of this injector (part of the planned disassembly and inspection - D&I - activity) revealed no anomalous conditions or incipient metallurgical problems. Thus, it is concluded that for a columbium alloy engine <u>no routine maintenance</u> is required prior to and after use.

During Phase I, an optical/photographic means of examining the injector was developed. Utilization of this aid assisted in diagnosis of an injector problem during Phase I. This same method was utilized throughout Phase III and appears adequate for incipient problem identification.

<u>Life</u>

During Phase I, a life of five missions* was demonstrated and, during Phase III a life of 10 missions* was successfully demonstrated. The D & I results from the 10-mission engine indicate the columbium alloy thrust chamber assembly (includes injector) to be in excellent condition. Thus, the 100-mission life requirement appears to require only demonstration. This also appears to be the case for the valve. Phase III D & I indicated the valve seat to be in excellent condition after 6400 hot fire cycles.

<u>Engine Performance and Operation</u>

The two-phase test program thoroughly evaluated engine operational and performance characteristics with and without helium-saturated propellants and at nominal and off design conditions. All performance and engine life objectives of the program were successfully demonstrated.

<u>Steady State Performance</u>

The steady-state program target of 295 lb-sec/lbm was demonstrated on two engines in Phase III. No performance impact due to helium saturation exists.

<u>Design Limits</u>

Off-design testing and adverse attitude tests (nozzle-up) were successfully demonstrated.

*Equivalent to 25 and 50 missions, respectively, as defined by RI/SD.

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Combustion Stability

Three engines successfully demonstrated the dynamic stability of the injector design by bomb tests. Recovery was always within the 20 ms requirement (1.2 ms maximum damp time). Tests were conducted during all critical operational times*, and with and without helium saturation of propellants.

High Altitude Ignition

Tests were conducted on three engines, with and without helium-saturated propellants over the projected operational mixture ratio and chamber pressure range, with no spike greater than 740 psia. Consequently, high altitude ignition was successfully demonstrated.

Pulse Mode Performance

Four engines were subjected to pulse mode testing with and without helium saturation for impulse bit and pulse specific impulse characterization. The pulse performance goals were demonstrated. A minimum pulse width of 30 ms was successfully demonstrated with high reliability and excellent repeatability. Although pulses less than 30 ms were demonstrated, there is a possible susceptibility to excessive engine cooling for very short pulses at low pulse densities. Helium saturation of propellants generally improves pulse performance characteristics.

Thermal Characteristics

Worst case mission duty cycle and maximum endurance tests were conducted with two engines during Phase III and one engine during Phase I. All units demonstrated external surface temperatures below the 800°F requirement. During both program test phases extensive testing was conducted to characterize the injector. These tests indicated the throat maximum operating insulated temperature was 2100°F at nominal operating conditions and 2300°F at worst case operating conditions.

*Start and shutdown transients as well as steady state, maximum operational requirements - high, nominal and low chamber pressures.

TABLE 2-1. ENGINE OPERATION AND GOALS

	PROGRAM GOALS	PROGRAM RESULTS
Vacuum Steady State Thrust (F_∞), lb.	600	600
Exhaust Nozzle Area Ratio (e)	40	40
Vacuum Specific Impulse (I_{sp})	295	295
Steady State Chamber Pressure (P_c), psia	200	200
Engine Steady State Mixture Ratio (O/F)	1.6	1.6
Propellants		
Oxidizer	N_2O_4 (MIL-P-26539C MON-1)	N_2O_4
Fuel	MMH (MIL-P-27404A)	MMH
Pressurant	Helium	Helium
Propellant Feed Conditions		
Static Pressure, psia	300 ± 6	300
Dynamic Pressure, psia	290 ± 10	290
Temperature, °F	75 ± 35	75 ± 35
Valve Voltage, VDC	28 ± 4	32
Minimum Impulse Bit (Nom. Conditions) lb-sec.	30 ± 10	26.9±1.2*
Maximum Pulse Frequency, cps	5	5
Maximum Single Firing, sec	600	600
Maximum Firing Time Per Mission, sec	1000	1009
Maximum Number Pulses per Mission	2000	120**
Pulsing Specific Impulse (Goal), sec	220	246
Engine Life**	100 missions	10 missions***
	10 Years	Not tested
	100,000 sec	10,400 sec**
	200,000 pulses	6,377 pulses
Stability		
High Frequency	Dynamically Stable - Recovery in 20 ms	Stable-Bomb tested over O/F & P_c band
Low Frequency	Oscillations - ±5%	Less than ±5%
Thrust Vector Alignment, degrees	0.5	Not tested
Maximum Outer Wall Temperature, °F	800	800
Engine Environment		
Temperature, °F	-20 to +300	+20 to +160**
Launch/Reentry Temperature, °F (5 min)	2000 at nozzle exit 1500 at throat	Not tested
Pressure	S/L to 10^{-13} torr	0.1 psia to 10^{-5} torr**
Acceleration (twice/mission)	3.5g for 1 min	Not tested
Vibration	Sinusoidal Vibration (6 min/mission) 5-23 Hz 1g 23-40 Hz 0.036 in D.A.	10 cycles without failure
	Random Vib. (1 min/axis/mission) 20- 90 Hz $0.1g^2$/Hz 90- 180 Hz +12 db/octave 180- 350 Hz $1.6g^2$/Hz 350-2000 Hz -6db/octave	23 cycles to failure-use of dampers as corrective action.
Shock	TBD	Not tested
Rain (per mission)	0.5 in./hr for 0.5 hr	Not tested
Sand (per mission)	140-mesh-500 FPM(4 hr)	10 Cycles
Salt Atmosphere (per mission)	Coastal areas @ 75°±20°F for 30 days	10 Cycles for 0.5 hr
Humidity (per mission)	0-100% RH for 30 days	10 Cycles for 8 hr.
Leakage of Propellants or Comb. Gases	None	None

* Electrical Pulse Width - EPW = 50 ms. Minimum impulse bits to 8 lb-sec were demonstrated.

** No intent to demonstrate due to program cost limitation.

*** Equivalent to 50 missions per RI/SD.

3.0 CONCLUSIONS/RECOMMENDATIONS

The Phase I and Phase III test programs have firmly established that the Bell vortex-cooled columbium alloy engine is capable of meeting the performance requirements of the Space Shuttle RCS application. Life and engine maintenance requirements appear to be achievable, but demonstration is required.

This technology program concerned itself primarily with engine (thrust chamber, injector, valve) technology since this is the heart of any rocket propulsion system. However, two auxiliary areas - flight instrumentation and valve durability (cycle life) - deserve further attention.

Flight instrumentation was not addressed at all during the technology program. This is a significant area because of the number of engines involved on a single craft, and potential instrument operational problems as well as instrument developmental problems. Thus, it is recommended that a two-part Analysis-Technology-Test program be conducted as follows:

Phase I - Requirements definition, instrument operation analysis, design selection and analysis

Phase II - Instrument design, fabrication, and test evaluation

The RCS propellant valve was only peripherally addressed during the technology program*. This is also an important area because of its functional requirements. Thus, some additional effort could be judiciously expended in a single-phase program to evaluate two valves for:

a. Cycle life

b. Seat and seat component wear

c. Material compatibility

d. Filter material requirements

e. Flexure sleeve dynamic requirements

*Bell subsequently successfully tested the torque motor valve through >200,000 cycles using simulated propellants with no leakage.

4.0 SPACE SHUTTLE RCS ENGINE TECHNOLOGY PROGRAM

4.1 Phase I

Phase I was the first iteration of test, analyses, and design leading to the fabrication of hardware during Phase II.

4.1.1 Test Program

The test program was intended to supplement analytical efforts by providing additional data defining engine performance. Testing was initiated in August of 1972 and continued through mid-December 1972. The testing, as presented in Figure 4.1-1, included the following general areas:

- Injector cold flow
- Helium saturation effects
- Combustion stability
- High-altitude ignition
- Thermal and mission duty cycles
- Pulse mode performance
- Post-fire methods
- Valve cycle life and firing
- Off-limits operation

A detailed listing of the fire tests performed is shown in Appendix II.

4.1.1.1 Test Hardware Descriptions

The test hardware used during the Phase I tests was designed to bolt together to allow for ease of test, flexibility in assembly/disassembly for inspection, and test interchangeability. A test engine consisted of a bipropellant valve, a prototype injector, one of two chambers, and one of two nozzle extensions. In addition, a thermal insulation blanket was installed for many of the tests.

Figure 4.1-2 shows a prototype engine assembly. Figure 4.1-3 shows hardware with the thermal insulation blanket.

1. Prototype Hardware

Injectors

The prototype injectors (see Figure 4.1-4) are of all-welded construction with 36 elements: 24 outer core doublets and 12 inner core doublets. They have a fuel vortex barrier fed by six injection orifices.

This selection (36 elements-unlike doublets) was made based on pressure evolution mapping, testing prior to contract award and BAC experience. The oxidizer streams are located so that they are injected inward away from the chamber wall to prevent oxidizer from flowing directly on the chamber wall should fuel orifice plugging occur (unlikely due to use of large orifices) to provide a large thermal margin and high reliability.

Bell's unique use of the vortex principle (fuel vortex barrier cooling) provides excellent uniformity (compared to fuel axial or doublet coolant injection) and high efficiency which allows for lower wall temperatures and higher performance. Fuel vortex barrier cooling is provided by tangential injection of fuel (see Figure 4.1-5). The liquid fuel flows over a vortex dam which together with drag from the core straightens the flow. Complete coverage is attained even with plugged barrier orifices. Only a few barrier orifices are required which enhances the contamination insensitivity of the design. The fuel vortex barrier provides predictable control of the thermal and gas species at the wall. Correlation of the coolant flow rate with chamber design characteristics is shown in Appendix I.

The orifice plate and injector flange are fabricated of a columbium alloy (Cb-1Zr). The orifices are inspected by visual inspection (backside and frontside) as well as cold flow prior to buildup to injector. The other parts (covers, tubes, support, etc.) of the injector are fabricated of a titanium alloy (6Al-4V) to minimize the heat soakback to the valve and propellants. The injector is all welded using electron beam welding and inspectable by radiographic and/or ultrasonic techniques with all welds and inspection techniques developed and demonstrated on the BAC Minuteman III PSRE program.

The valve interface can be adapted to take either a test valve or the prototype valve which is bolted to the injector and sealed with teflon coated metal seals.

The prototype injectors are provided with flange and seal surfaces to mate with the test combustion chambers.

Dynamic stability is accomplished via acoustic cavities which are curved slots cut into the injector face for passive damping of acoustic combustion disturbances. The frequencies are as follows:

First Tangential	- 7530 Hz
First Radial	- 15,670 Hz
First Longitudinal	- 10,776 Hz

S/N DD-PR-2

Figure 4.1-6 shows the DD-PR type injector. This injector differs from the RDV or flight-type design in that no attempt was made to reduce the manifold dribble volumes.

Injector S/N DD-PR-2 was built as a backup for Phase I and the majority of testing was performed with this unit.

S/N RDV-DD-PR-1

The reduced dribble volume type injector incorporates flight-type propellant manifolds which minimize the volume downstream of the valve interface. However, manifold velocity control is maintained to promote uniform core stream characteristics. The reduced dribble volume improves the pulse mode characteristics of the design.

Other differences from the DD-PR design include a refinement of the valve mounting plate support and the elimination of a weld on the injector face.

The injection scheme is the same as in the DD-PR type injector.

The S/N RDV-DD-PR-1 injector (shown in Figure 4.1-7) was designed in Phase I and was tested early in Phase I. Due to a design deficiency in this injector, the bulk of testing in Phase I was conducted with S/N DD-PR-2.

S/N RDV-DD-PR-2

The second RDV-type injector (Figure 4.1-8) was built incorporating several changes from the initial RDV-type injector.

The fuel barrier percentage was reduced. The structural integrity was improved in a deficient area. The fuel inlet tube was modified to reduce the fuel pressure drop of the injector. Concentric injector face dams, as shown in Figure 4.1-8, were added to minimize the potential for manifold explosions, by preventing the fuel from entering the oxidizer manifold. This injector was tested at the end of the Phase I test program.

Prototype Bipropellant Valve

The prototype valve is a normally closed, torque motor operated, bipropellant valve manufactured by Moog, Inc. This valve was used for some of the Phase I hot fire tests. (See Figure 4.1-9)

The valve consists of a fuel propellant chamber, an oxidizer propellant chamber, and a torque motor. The fuel and oxidizer flapper buttons are attached to the armature of the torque motor by means of flapper rods. The flapper rods are supported by flexure tubes which act as flexible fluid barriers between the electromagnetic (torque motor) and propellant chambers of the valve.

Combustion Chambers

Two prototype combustion chambers were used during the Phase I tests. They are identical except for the silicide coatings used.

The chambers, one of which is shown in Figure 4.1-2 consist of a columbium (SCb-291) liner with columbium (Cb-1Zr) flanges. They are coated inside and outside with a silicide coating which is 0.002 to 0.005 inch thick. Each is provided with flanges which bolt to the injector at the upstream end and to a nozzle extension on the downstream end.

Chamber S/N P-1 is coated with GTE-Sylvania (now HiTempCo) R512A silicide coating. Chamber S/N P-2 (E) is coated with the same company's R512E coating.

The combustion chamber design has a characteristic length of 14 inches with a contraction ratio of 5.5 with a 3.9 inch length from the injector face to throat.

Nozzle Extensions

Two different nozzle extensions were used.

ϵ = 31 Extension (See Figure 4.1-2)

This unit was used during the engine tests in the B-2 simulated altitude facility. It consists of an SCb-291 nozzle section with a Cb-1Zr flange welded to it. The assembly is coated inside and out with R508C silicide coating and is designed to bolt to the columbium thrust chambers.

ϵ = 15 Extension

This unit is similar to the ϵ = 31 extension except that it is shorter, is made of C103 instead of SCb-291, and is coated with R512E silicide coating. It was used in the sea-level cell in conjunction with the simulated altitude duct.

Thermal Blanket

During much of the testing with the columbium chamber and nozzle extension, the outer columbium surfaces of the engine were covered by a thermal insulation blanket of Dynaflex (6 lb/ft^3 density) with an outer covering of 0.005 inch stainless foil. (See Figure 4.1-3)

2. Test Hardware

To facilitate certain aspects of the test program, the following items of non-prototype hardware were used.

Building Block Injector

The off-design tests were done on the "building block" injector used earlier in the Bell Aerospace program. This injector has separately fed core (primary) and barrier elements. The injector used (DD-BB), has core elements arranged in the same pattern as the prototype injector. The core hole sizes and resulting pressure drops are similar.

Test Valves

One of two non-prototype propellant valves was used for the majority of the Phase I tests. These are normally open poppet-type bipropellant valves.

The valve, shown in Figure 4.1-10, consists of an activating piston chamber with a spring-loaded piston, an oxidizer propellant chamber, and a fuel propellant chamber. All three chambers are enclosed in a single body with the activating piston chamber being in between the two propellant chambers. The activating piston shaft, the oxidizer poppet shaft, and the fuel poppet shaft are interconnected externally by a single yoke.

The original valve was modified by adding a manifold so that the valve could be mounted directly to the injector and to keep the residual volume at a minimum.

Bomb Chamber

The bomb chamber (Figure 4.1-11) is identical in interior contour to the columbium chamber, but has essentially no expansion beyond the throat. It is constructed of stainless steel and has ports for one flush-mounted and two water-cooled Kistler pressure transducers located 90° from each other to detect the tangential mode (most probable of combustion instability). It also has a port for the insertion of the "bombs" and provision for the installation of accelerometers.

This chamber was used for some sea-level checkout firings, the stability series, and the high-altitude ignition tests.

3. Test Facilities

Sea-Level Facility (D-3)

Test cell D-3 is basically an open-air sea-level facility in which the engine fires horizontally. Additional capabilities were added through the use of two different bolt-on ducts as shown in Figures 4.1-12 and 4.1-13. High-altitude ignition tests were performed in D-3 using a chamber equipped with a high-vacuum pump. Long duration simulated altitude tests were performed using a self-aspirating diffuser duct.

Altitude Cell (B-2)

Test Cell B-2 is a 12-ft diameter, 18-ft high vacuum chamber in which simulated altitudes of approximately 100,000 ft. can be maintained during 30-sec. steady-state firings as shown in Figure 4.1-14.

This test stand is instrumented to measure thrust and was equipped with a Propellant Metering System (PMS) for the Phase I tests.

The PMS is a small propellant feed system in which the tanks may be repeatedly filled to exactly the same level. The amount of propellant required to refill to this level can be precisely measured. This system allows the average specific impulse and mixture ratio for a train of pulses to be determined.

4. Data Acquisition and Handling

Digital Data

The majority of test parameters necessary for calculating engine firing performance was recorded on one of two types of high-speed Beckman analog-to-digital conversion systems: 6.9KHz for sea-level testing and duration firings in the sea-level facility, and 30KHz for altitude testing, pulse mode evaluation and Mission Duty Cycles (MDC's). These systems differ from one another in sampling rates and method of recording digitized data, but have the same basic accuracy. These data were then processed by an IBM 360/44 general-purpose digital computer, using programs developed by Bell Aerospace.

Analog Data

Analog data were used to assess transient behavior of the engine such as combustion stability and ignition characteristics.

For transducer outputs having high-frequency components, a wider bandwidth is required than can presently be handled directly by the digital acquisition systems. In such instances, the raw data were converted to a wideband FM signal and recorded on magnetic tape via one of two Sangamo analog tape machines.

4.1.1.2 Testing

Phase I testing was performed in accordance with the test matrix (Figure 4.1-1). One prototype, two flight-type injectors, and one building block injector, were utilized. These are identified respectively as: S/N DD-PR-2, S/N RDV-DD-PR-1, S/N RDV-DD-PR-2, and S/N DD-BB. All test results are presented in Appendix II.

The test program was initiated with injector S/N RDV-DD-PR-1. This injector was cold-flowed, acceptance-tested (with performance characterization), and then utilized in bomb stability and high-altitude ignition tests.

The stability test series consisted of 25 bomb tests of various durations from 0.25 to 1.5 seconds. Detonation was electrical at various times throughout the firings(i.e., during start transients and during steady-state). The bombs yielded overpressures in the 200% to 300% range. All tests were conducted with helium-saturated propellants. Typical damping times were 1 ms with the maximum 1.2 ms with no disturbances sustained. Consequently, dynamic stability was demonstrated.

The high altitude test series consisted of 20-50 ms firings at simulated altitude in excess of 250,000 ft. The maximum spike level recorded was 398 psia. The maximum spike duration was 2.5 ms. Again, dynamic stability was demonstrated.

Additional planned tests were not conducted due to the low level of spikes.

The high altitude start series was performed in Test Cell D-3 (sea level cell) utilizing a small vacuum pump simulated altitude chamber as shown in Figure 4.1-13.

Test procedure consisted of pumping the duct down to the required simulated altitude (250,000 ft) and then firing the engine for the required duration.

Upon completion of the high altitude test series, it was apparent that a structural failure had occurred in the injector. The design deficiency was traced to a sharp radius in the fuel manifold.

Analysis of the failure indicated some freezing of oxidizer due to test facility limitations which allowed some residual fuel to dribble into the oxidizer manifold causing a reaction. The minimum section between the oxidizer and fuel manifold locally failed due to the reaction. This deficiency was caused by an inaccurate assessment of the stress concentration factors. The corrective action (implemented on all subsequent injectors) was to maximize the section between the oxidizer and fuel manifolds and increase the radius. This significantly reduced the stress concentration factors as well as incorporate face dams between propellant orifice rows to prevent cross flow (see Figure 4.1-8).

Injector S/N DD-PR-2 was assembled (See Figure 4.1-15) into a thrust chamber, as a substitute for S/N RDV-DD-PR-1, and was utilized for assessing the pulse performance, mission profile, and durability testing.

This testing utilized saturated and unsaturated propellants and consisted of the firing of the engine through mission profiles A, B, C, D, and E. Test results from these mission profiles, presented in Appendix II, completely define the operational characteristics of the engine configuration.

Mission Profile A was a four part test series consisting of the following:

1. Mixture ratio and P_c - 5-point text matrix for steady state performance definition.

2. Thermal Duty Cycle (see Table 4.1-1) consisted of 10-10 second and 40-50 ms firings with long heat soak periods to impose the worst case thermal conditions on the injector. (102 seconds on-time over a 21 minute elapsed duration).

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THERMAL DUTY CYCLE

NO. PULSES	ON-TIME	OFF TIME
1	10 SEC.	1 MIN.
1	10 SEC.	3 MIN.
1	10 SEC.	3 MIN.
1	10 SEC.	4 MIN.
10	50 MS	150 MS
1	10 SEC.	4 MIN.
10	50 MS	450 MS
1	10 SEC.	3 MIN.
10	50 MS	950 MS
1	10 SEC.	2 MIN.
10	50 MS	

TABLE 4.1-1

3. Pulse performance versus width and frequency series.

4. A worst case mission duty cycle (see Table 4.1-2) consisted of 10-100 second and 110 pulses of 50, 90 and 170 ms firings interspersed. (1009 seconds on time over a 25 minute elapsed duration)

Mission Profile B was primarily a continuous 600 second firing preceded by a pulse heating series with pulse widths of 50, 90 and 170 ms.

Mission Profile C was identical to A except the propellants were helium saturated.

Mission Profile D was a two part test series intended to do the following:

A. Establish pulse performance as a function of propellant temperature from 40°F to 110°F.

B. Determine performance and thermal shifts for the worst case conditions at high O/F, high P_c and propellant temperatures of 110°F.

Mission Profile E was identical to Mission Profile D except the propellants were saturated.

Representative steady state test results from Mission Profiles A, B and C are presented in Figures 4.1-16, 4.1-17 and 4.1-18. There were some propellant feed system regulator problems which resulted in more than normal mixture ratio, P_c and thrust variability but the fact that I_{sp} is invariant is proof that the engine performance was stable during these critical duty cycles. The maximum throat temperature was 2165°F.

Representative pulse performance test results from Mission Profiles A, C and E are presented in Figures 4.1-19 through 4.1-22. Pulse impulse versus frequency (Figure 4.1-19) indicates impulse bit is a strong positive function of pulse frequency which would be expected based on manifold dribble/propellant evaporation considerations. Impulse bit as a function of EPW (Figure 4.1-20) also shows a positive function with pulse width but no dependence on helium saturation. The pulse performance as a function of pulse width and frequency (Figure 4.1-21) shows results consistant with the impulse bit results. Figure 4.1-22 shows the extension of the pulse performance data to the flight type Phase III engine.

TABLE 4.1-2

WORST CASE MISSION DUTY CYCLE

NO. PULSES	ON-TIME	FREQUENCY	OFF-TIME
10	50 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	50 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	90 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	50 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	170 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	50 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	90 MS	1 CPS	10 SEC.
	HEAT SOAK PERIOD		480 SEC.
1	100 SEC.	-	10 SEC.
10	50 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	170 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	50 MS	1 CPS	10 SEC.
1	100 SEC.	-	10 SEC.
10	90 MS	1 CPS	10 SEC.

As indicated above, this part of the test program produced results as expected except for a seat leakage problem which developed in the first Moog valve and a test valve seat failure. The test valve had been used for several years and, when the Kel-F seat broke up due to excessive cycling and aging, the injector became contaminated with the seat material causing erratic injection for several orifices resulting in a small burn through at the throat of the thrust chamber. Since the test valve is completely different from the flight-type valve, this problem is not relative to the flight engine.

The Moog valve seat leakage problem was found to be the result of reduced loading on the downstream seat. The valve was repaired and returned to test.

In order to obtain pulse performance data with a reduced dribble volume injector, injector S/N RDV-DD-PR-2 was fabricated and built into an engine assembly. Tests on this unit consisted of a series of steady-state checkout firings and pulse trains at 1 cps with pulse widths of 35, 50, 90, and 170 ms with ambient unsaturated propellants. This effort completed the prototype testing of Phase I.

Throughout Phase I, off-design tests were conducted with the building block injector S/N DD-BB. The test objective was to determine injector sensitivity to orifice plugging. These tests consisted of successively plugging selected injector orifices; then conducting steady-state performance and thermal evaluation tests. Up to 33% of adjacent barrier orifices were plugged, with the only results being generally increased thrust chamber temperatures resulting from the loss of barrier flow. In addition, two adjacent outer primary fuel orifices were plugged, with a minor increase in wall temperatures.

In addition to the engine tests, three Moog valves were tested during Phase I. Two were prototype valves and one was a modified Minuteman III valve (same motor, flexure sleeve and body but different seat configuration). The Minuteman valve was used to obtain heat transfer data.

Prototype valve S/N-1 was subjected to various bench tests at both Moog and Bell and was fired as noted previously. Prototype valve S/N-2 was bench tested at Moog for endurance cycle testing.

4.1.1.3 Summary and Conclusions

The Phase I testing demonstrated steady-state and pulse-mode performance, dynamic combustion stability, reusability without servicing, and various duty cycles including worst case and duration firings. As a result, the goals of the RCS Technology Program are considered feasible and practical.

Table 4.1-3 presents the engine performance goals and the results obtained from Phase I testing. In addition to the goals listed in the table, the Phase I test program evaluated high-altitude ignition effects and maintenance and inspection requirements.

A complete summary of all test data is presented in Appendix II.

<u>Steady State Performance</u>

The engine design demonstrated performance and thermal characteristics as presented in Figures 4.1-23 through 4.1-27, which indicates the performance goals could be achieved with a flight type engine design. In addition it was found that helium saturated propellants do not impact steady state performance. The results of the correlations are shown in Figures 4.1-28 through 4.1-30 over mixture ratio, chamber pressure and propellant temperature ranges.

<u>Combustion Stability</u>

Tests were conducted on the reduced dribble volume (RDV) injector with helium saturated propellants to evaluate the stability of the engine at varying operating conditions including start transients. The results indicate the injector is dynamically stable, based on recovery times of 1 ms in every case.

<u>High Altitude Ignition Tests</u>

The RDV injector was evaluated for ignition spike characteristics at pressure altitudes of 250,000 feet greater, using low temperature propellants. These tests evaluated normal as well as helium-saturated propellants. All ignition spikes were less than 400 psia. It is concluded that there is no ignition spike problem for the Bell RCS injector concept.

<u>Pulse Mode Performance</u>

Pulse performance was obtained on two different injectors. Results indicate that increased chamber pressure, propellant feed temperature, and electrical pulse width (EPW) and frequency increase both pulse specific impulse and pulse total impulse as shown in Figures 4.1-31 through 4.1-33.

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TABLE 4.1-3

ENGINE PERFORMANCE GOALS AND PHASE I RESULTS

PARAMETER	PROJECT GOAL	PHASE I RESULT	COMMENT
Steady State			
Vacuum Specific Impulse (ε = 40)	295 lbf-sec/lbm	295	
Maximum Single Burn	600 sec	Demonstrated	
Thrust Vector Alignment	±0.5°	Not tested	
Pulse Mode			
Minimum Impulse Bit (EPW - 50 ms)	30±10 lbf-sec	26.9 (ε = 40, W/He)	7.5 lbf-sec @ EPW = 30 ms
Vacuum Specific Impulse	220 lbf/lbm/sec	236 @ 5 cps, 205 @ 1 cps } ε - 40 W/He	
Maximum Pulse Frequency	5 cps	Demonstrated	1000 pulses @ EPW = 50 ms in single train.
Stability			
High-Frequency	20 ms recovery from bomb	Demonstrated	Recovery times 1 ms
Low-Frequency	±5% max.	No problem	
Feed Condition Extremes			
(Includes helium-saturated propellants)	Meet goals using same	Demonstrated	
Maximum Outer Wall Temperature			
(Insulated)	800°F	Can achieve	
Engine Environment			
(Vibration, temperature, shock, rain, etc)	Survive	Not tested	
Leakage			
Propellant (Valve/interface)	None	Interface no problem - Seat Modification required	
Combustion Gases (Combustion Chamber)	None	See engine life	
Engine Life	100 missions		
Thrust Chamber			
Firing Time	1000 sec/mission	Single injector: 3.8 missions; Single chamber: 5.0 missions	No injector service: 3.2 missions
Firing Cycles	2000/mission	Single injector: 3.7 missions; Single chamber: 3.8 missions	3.4 missions
Bipropellant Valve	2000 cycles/mission	7.5 missions	

Mixture ratio has little or no impact. Helium saturation increases the pulse specific impulse and pulse total impulse at frequencies of 1 cps or less, but may have a negative effect at a frequency of 5 cps as shown in Figure 4.1-34.

The RDV flight-type injector gives better performance than the prototype injector at every tested condition. The improvement was 18.5% in pulse total impulse and 16% in pulse specific impulse at electrical pulse widths of 50 ms and a frequency of 1 cps as shown in Figure 4.1-33.

Isolated pulses were found to have characteristics similar to those at 1 cps. Data also indicate that pulses of 1 cps at the same EPW will have good reproducibility (range of total impulse: 2 lbf-sec) as shown in Figure 4.1-35. Helium saturation produces more uniform results but pulsing frequencies greater than 1 cps increase the variability.

Moog Prototype Bipropellant Valve

Two Moog prototype RCS bipropellant valves were tested during Phase I. Both experienced seat leakage due to cold flow of the teflon downstream oxidizer seat. However, one valve demonstrated 15,000 cycles (7.5 missions)* prior to leakage.

Design refinements incorporated into the Phase III valves demonstrated solution to the seat leakage problem.

Maintenance and Inspection

One injector/thrust chamber assembly was hot fire tested to approximately 4 space shuttle missions** without any flushing or maintenance. It is concluded that no post-flight maintenance of the RCS engine is required. A visual inspection using appropriate optical aids is adequate to verify flight-readiness of the injector and chamber.

Life

Phase I testing demonstrated the capability of the RCS engine to perform the equivalent of several space shuttle missions including duration burns, pulsing, and many duty cycles including the worst case duty cycle. These tests indicate that the Space Shuttle requirements can be met; however, demonstration is required.

*Equivalent to 30 missions as defined by RI/SD.

**Twenty missions as defined by RI/SD Space Shuttle Requirements.

4.1.2 Analyses

4.1.2.1 Systems Analyses

The RCS engine must meet the conditions developed by the system. System analyses were conducted to define the variations representative of the final RCS engine application. The considerations assessed were:

- Baseline system (forward module - 16 engines used).
- Engine/system interface pressures.
- Off-rated performance conditions.
- Multiple engine operation.

The analyses were based on the nominal case of two engines firing off a given line. The worst case is based on one engine or four engines firing off a given line. The feed pressure variations fall within the range of 280-300 psia, with a maximum mismatch of 3 psi. In addition, a mismatch of 15°F in propellant temperature was combined with these feed pressure effects to estimate the total excursions in thrust and mixture ratio. The worst case results indicate a change in mixture ratio of $^{+2.8}_{-2.9}$% and a change in thrust of ±3.3%. Appendix III shows the curves and details of these analyses.

An analysis was conducted of the off-rated engine feed pressure schedule and tolerance bands for a secondary regulation system and for a typical relief valve. The results are as follows:

Condition	Maximum Feed Pressure - psia
Primary Regulation	299
Secondary Regulation	309
Relief Valve Control	
Reseat	315.5
Min. Crack	330.5
Max. Crack	340.5
Reg. Outlet Pressure (max. flow)	355.5
Long Term Coast	340.5

Off-rated conditions are normally encountered only when the primary pressurization system malfunctions and pressure control shifts first to the secondary regulator and eventually to the relief valve with the corresponding feed pressures shown above.

In addition, analysis showed that tank pressures above the primary regulation system level can be achieved as a result of an extended coast period wherein heating of the propellant and pressurant occurs. This can create an engine feed pressure condition as high as the maximum relief valve cracking pressure.

4.1.2.2 Thermal Analyses

The thermal analyses of the flight type engine to be tested in Phase III was completed. Appendix IV shows the temperatures of various parts of the engine as a function of duty cycle. The propellant resultant temperatures at the injector manifold for the Phase I engine are shown as follows:

Propellant	Inlet Temp-^oF	Resultant max. Outlet Temp-^oF	Sat. Temp-^oF
Oxidizer	70	119.4	190
Fuel	70	102.8	380
Oxidizer	110	158	190
Fuel	110	141	380

Flight thermal effects were also investigated. The engine was analyzed for heat soak due to reentry temperatures of 1500^oF at the throat. The valve temperature (initially 80^oF) is increased by 0.2^oF. The injector temperature reaches a maximum of 278^oF at the fuel dam. The dam temperature is 283^oF due to steady state operation, but reaches 550^oF during a 0.01 duty cycle (on time/total time).

Low temperature flight thermal effects were also analyzed. Based on the injector being at -20^oF, a total of 1.7 watts (heater power) is required to maintain the valve at +20^oF.

In addition, the ZOT phenomena was also investigated for 27 and 50 ms pulses. Assuming 25% of the N_2O_4 in the dribble volume of the injector is vaporized, the amount of evaporative cooling is 0.6^oF per 50 ms firing or 3.6^oF per 27 ms firing. Consequently, it would take 50 pulses of 50 ms each to freeze the oxidizer based on a starting point of 40^oF ignoring radiation effects from the environment. The injector is designed with face dams (Figure 4.1-8) to prevent fuel from getting into the oxidizer orifices/manifold.

The McDonnell-Douglas CONTAM-TCC program indicates a maximum spike of 300 psia due to hydrazinium nitrate reactions as the fuel barrier and inward canting oxidizer prevent serious hydrazinium nitrate production.

4.1.2.3 Computer Simulation (CONTAM-TCC)*

The approach taken to assess the capability of the TCC subprogram to model the RCS engine involved a simulation of an engine of known configuration firing under known test conditions. The configuration consisted of injector S/N DD-PR-2 with the test valve tested at sea level at nominal conditions with close coupled, high-response chamber pressure instrumentation. Figure 4.1-36 shows the comparison of the analytical prediction with the test data. The analytical prediction was adjusted by 3 ms to account for the pumping action of the test valve which is not modeled. The agreement for the first 34 ms is well within the error band due to uncertainty in propellant temperature which is substantial at sea level. The two step chamber pressure rise instead of the three observed is probably due to the analytical assumption that the inner and outer manifold rings prime simultaneously which is not correct based on hydraulic analysis and test. The different rate of pressure decay after the maximum in chamber pressure is associated with a pressure wave in the propellant feed lines which is not modeled in the lump parameter TCC flow algorithm. The predicted steady-state chamber pressure variation is due to improper modeling of the vortex barrier flow and its combustion at the time the simulation was made which has since been corrected.

Figure 4.1-37 shows the simulated 30 lb-sec. minimum impulse bit (requirements minimum) with the Phase III engine at altitude at nominal conditions with an anticipated vehicle feed system. It represents a first pulse with completely unprimed manifolds.

Pulse performance and exhaust product composition were analyzed using the latest version of the TCC computer program which includes the proper simulation of the fuel vortex barrier. The analyses are based on a first 30 lb-sec impulse bit at nominal conditions with unprimed injector which constitutes a worst case condition. Correlation of the computer analysis with the hot fire test results are shown in Figure 4.1-38. The exhaust product composition into the nozzle is 1.5% fuel retained as fuel film and 1.3% retained as other than fuel film. It is anticipated that the residual fuel film will be uniformly distributed in the nozzle where it will be burned off in the subsequent pulse. The computer program does not calculate the unburned propellants generated overboard through the plume which is significantly less.

*A computer program developed by McDonnell-Douglas Astronautics Company which quantatively describes the detailed physical processes occurring during the transient and steady-state operation of bipropellant rocket engines.

The Phase III engine (with Moog valve) was analyzed for determining the formation of nitrates on a cold engine wall. The results show essentially no nitrate is formed during start and firing. After shutdown, nitrates are formed and reach a peak about 98 ms after the start of the firing. The peak explosion pressure is 354 psia. After this time the amount of nitrate decreases as fuel on the wall washes it out of the thrust chamber. Preliminary assessments indicate little change on pulses to 27 ms (380 psia - peak explosion pressure).

4.1.2.4 <u>Hydraulic Analysis</u>

A complete hydraulic analysis was conducted on the Phase III engine. The results indicate the uniformity on the oxidizer and fuel sides in the core is ±1%. The uniformity in core mixture ratio is $^{+1.5}_{-2.2}\%$ for both inner and outer ring of core doublets.

4.1.2.5 <u>Reliability</u>

A failure mode and effects analysis was conducted as shown in Appendix V. The assumptions utilized to perform these analyses are:

(1) The propellant feed system will include means of isolating individual engines from the propellant manifolds.

(2) The bipropellant valves will be individually commandable which allows any engine to be electrically disabled.

(3) Performance monitoring of individual engines (i.e., chamber pressure, electrical verification of valve operation) is possible.

(4) Ground servicing and inspection is minimal.

The analysis defines failure mode, cause, classification, symptoms, action, effects (engine/flight hardware, mission and ground handling personnel and procedures), prevention (design features, acceptance test, and post-flight service/inspection). Classification of failures is defined as follows:

Critical - Failure causing mission abort or safety hazard.

Major - Failure degrading reliability or performance of the system.

Minor - Failure having no significant effect on reliability or performance.

4.1.2.6 Materials Selection

The RCS engine design requires a materials selection consistant with the requirements of long life (100 missions), safety and high reliability while providing high performance. This requires large margin with respect to melting point, fatigue life, creep resistance, minimization of material degradation, erosion resistance, ductility preservation and resistance to all corrosive environments which may be encountered.

The columbium injector (Cb-1Zr) was selected based on its excellent resistance to all environments and excellent producibility (over 6500 Cb-1Zr injectors produced on MM III program). The titanium (6Al-4V) alloy was selected for manifold covers, feed tubes and support based on its high strength, low thermal conductivity to minimize conduction to valve/propellants, resistance to all environments and excellent producibility (same as MM III - Cb-1Zr/Ti-6Al-4V injectors).

The thrust chamber and injector were selected as an all welded integral design for high reliability and minimization of areas for propellant accumulation. The thrust chamber (SCb-291/R512E) was selected based on its excellent resistance to all environments, large thermal margin and excellent producibility (over 6500 SCb-291 thrust chambers produced on MM III program).

The external insulation was selected as Dynaflex (12 lb/ft^3 density) based on its compatibility with the thrust chamber and environments.

The valve, with a 304L body and 17-7 flexure sleeves, was selected for strength and compatibility, titanium motor cap for compatibility and teflon seats for life and compatibility which is the same as MM III (over 7200 valves produced).

4.1.2.7 Structural Analyses

Structural analyses were conducted based on the thermal analyses previously conducted. Three major areas were investigated. Steady state structural analysis (Table 4.1-4) indicated large thermal margins. Weld joint fatigue analysis (Table 4.1-5) indicated large margins (14:1 minimum) to meet the life requirements of 10,000 thermal cycles (100,000 seconds). The engine also has a large capability to withstand spikes as shown in Table 4.1-6.

TABLE 4.1-4

STEADY STATE STRUCTURAL ANALYSIS

COMPONENT	STRESS TYPE	MARGIN OF SAFETY
Oxidizer/Fuel Inlet Tubes	Bending + Tensile at Weld Due to Pressure.	+9.40
Oxidizer Manifold Cover	Shear at Weld Joint Due to Pressure.	+9.00
Support	Bending Due to Vibrational Loading of Valve and Structure.	+24.60
Chamber	Stress Rupture for 50 Hours Life	+1.055

TABLE 4.1-5

WELD JOINT FATIGUE CAPABILITY

WELD JOINT NO.		MAX. TEMP. °F	MAX. STRESS KSI	THERMAL CYCLE CAPABILITY
1	(Tubes to valve mount)	80	1.5	10^7
2	(Fuel tube to orifice plate)	80	3.2	10^7
3	(Ox manifold to orifice plate inner weld)	80	6.6	10^7
4	(Ox manifold to orifice plate outer weld)	250	35.0	1.45×10^5
5	(Orifice plate to support)	450	9.8	10^7
6	(Support to valve mount)	90	2.6	10^7
7	(Orifice plate to vortex dam)	425	13.9	3×10^6
8	(Injector to chamber)	250	25.5	4×10^3

TABLE 4.1-6

SPIKE PRESSURE CAPABILITY

COMPONENT	STRESS TYPE	SPIKE CAPABILITY-PSI
OXIDIZER/FUEL INLET TUBES	TENSILE HOOP	18,400
OXIDIZER MANIFOLD COVER	SHEAR AT WELD JOINT	7,750
SUPPORT		UNAFFECTED
CHAMBER	TENSILE HOOP	4,630

4.1.2.8 Safety Considerations

The safety considerations in using a bipropellant RCS engine for Space Shuttle were briefly examined to identify specific design constraints on the engine and to support the maintenance and servicing considerations. Personnel safety in flight is inherently tied to the achievement of functional reliability and resulted in the design of the engine to operate at all known malfunction conditions. The hypergolic character of the N_2O_4/MMH propellant combination precludes large concentrations of partially mixed propellants and neither is susceptible to explosive decomposition.

These propellants are hazardous in air in relatively low concentrations; consequently, adequate precautions and decontamination procedures are necessary for personnel safety since the reusability aspects of the engine do not require decontamination after use. These propellants are storable liquids of relatively low vapor pressure and with a large experience base in handling the propellants and servicing systems containing these propellants. No new handling procedures are deemed necessary.

PHASE I TEST MATRIX

	TEST ASSEMBLIES			
TEST OBJECTIVES	RDV-DD -PR-1	DD-PR-2	RDV-DD -PR-2	DD-BB
Injector Cold Flow	X	X	X	-
Steaay State Performance	X	X	X	X
Off-Design	-	-	-	X
Helium Saturation	X	X	-	-
Combustion Stability	X	-	-	-
High-Altitude Ignition	X	-	-	-
Pulse Performance	-	X	X	-
Mission Duty Cycle	-	X	-	-
Durability	-	X	-	-
Post-Fire Servicing	None	None	None	As Req.

Figure 4.1-1

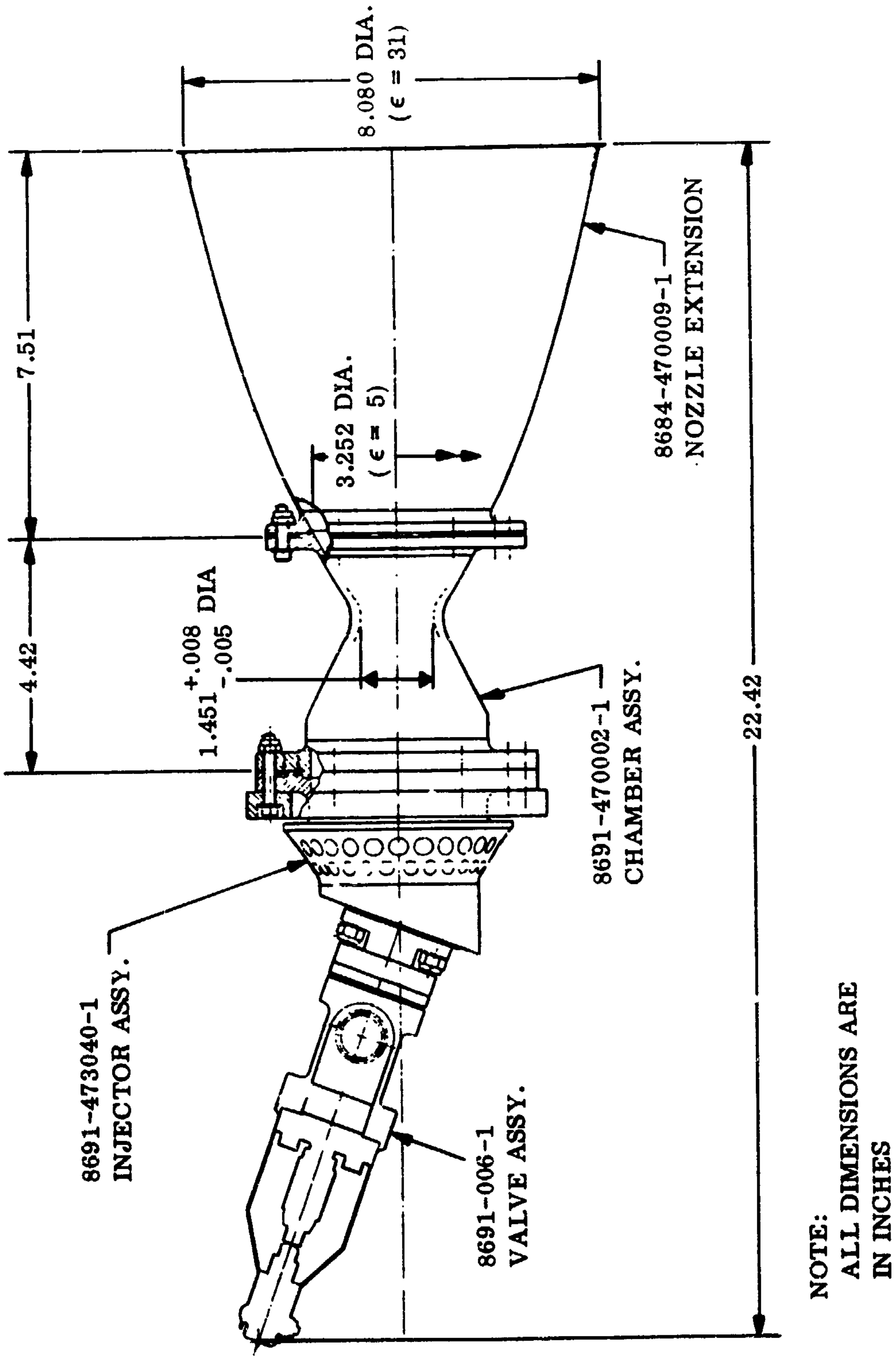


Figure 4.1-2 Prototype Engine (With Test Valve)

NOTE:
ALL DIMENSIONS ARE
IN INCHES

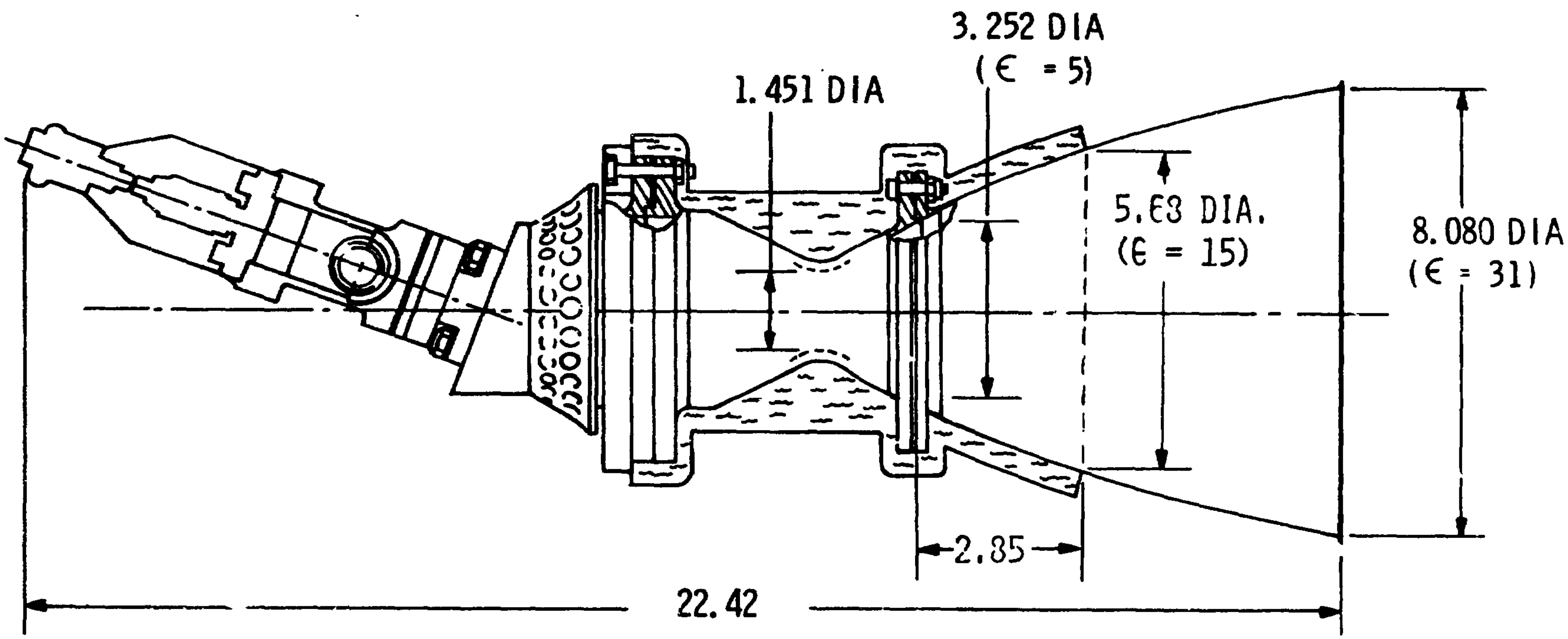


Figure 4.1-3 Prototype Engine (With Test Valve) and Thermal Insulation

INJECTOR FUEL FLOW PATTERN

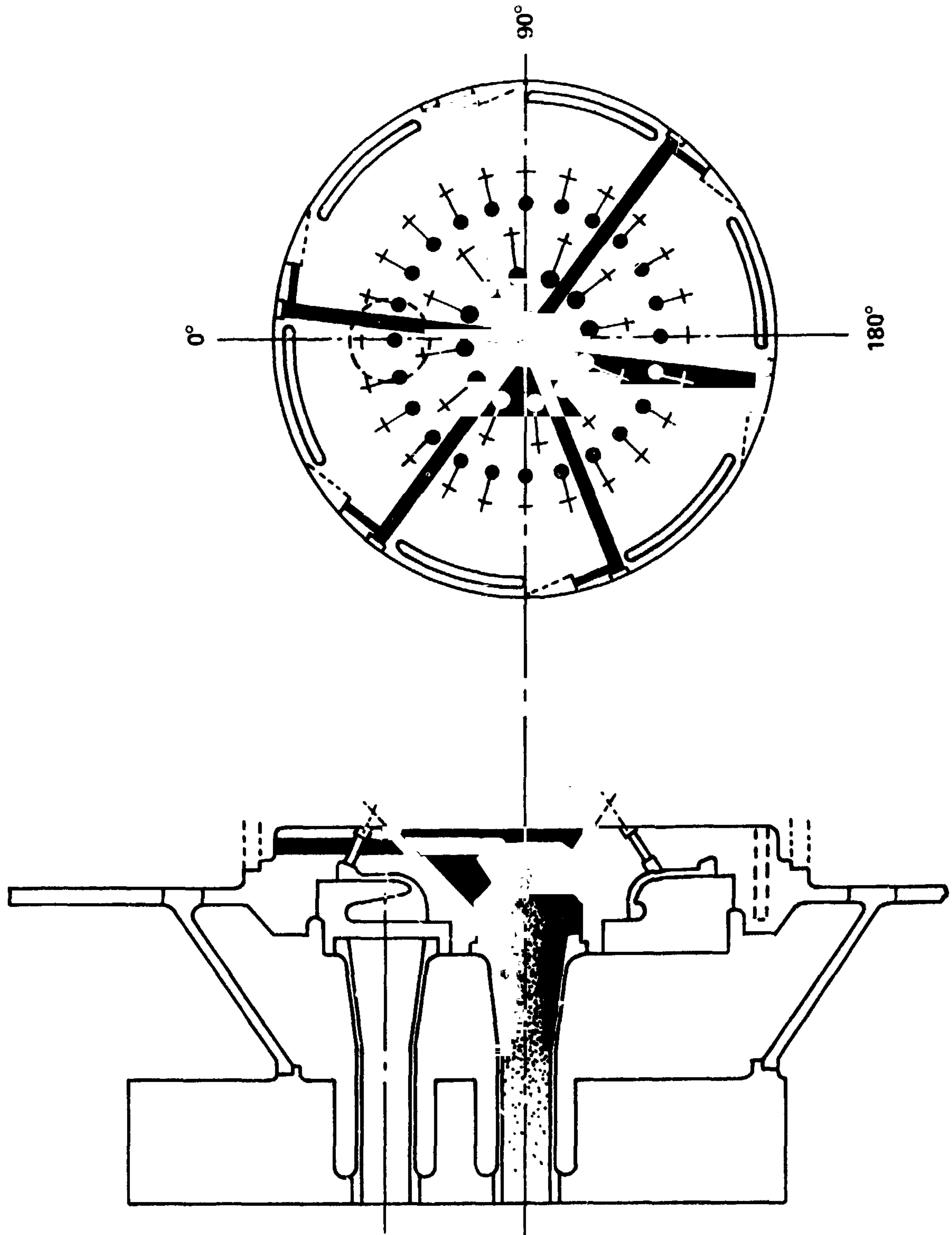
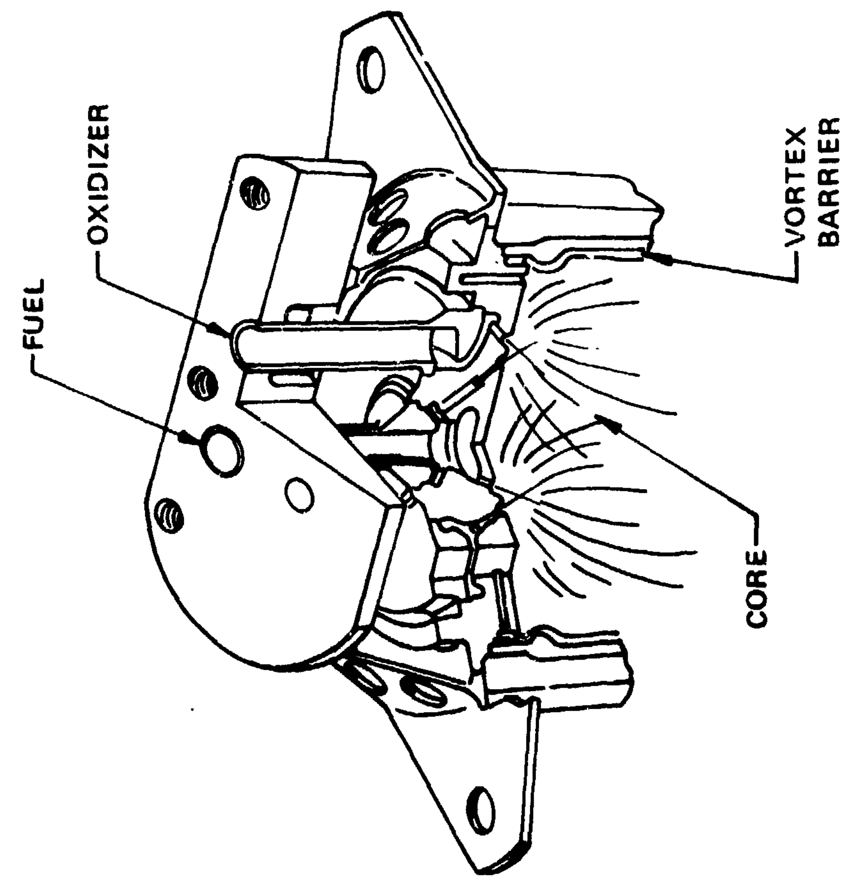


FIGURE 4.1-4



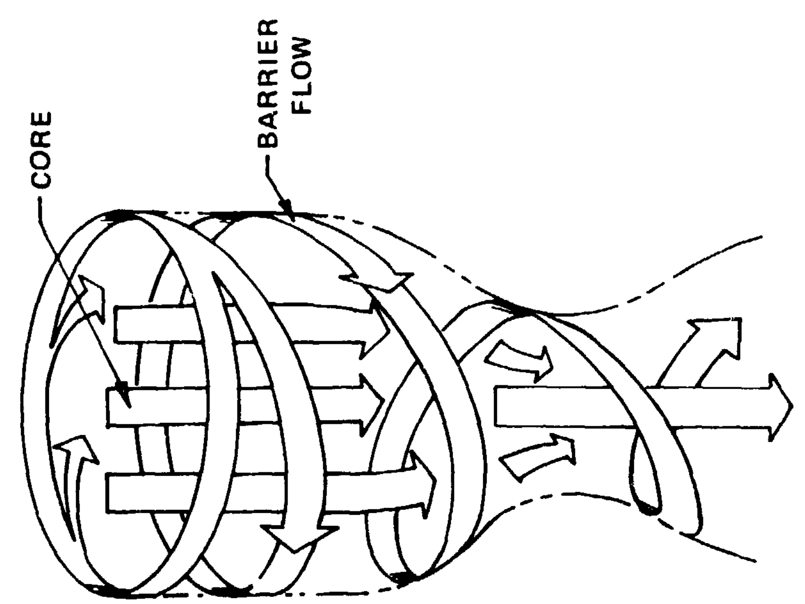


FIGURE 4.1-5 VORTEX COOLED BARRIER INJECTOR

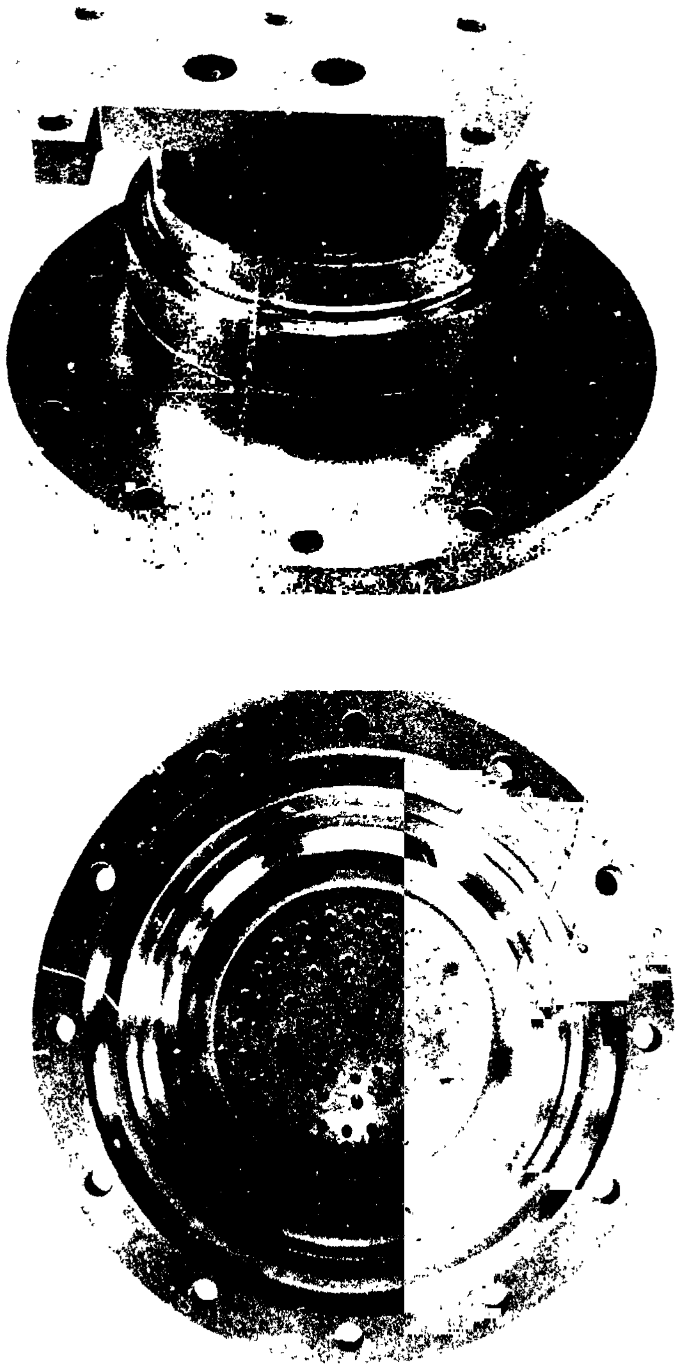


Figure 4.1-6 DD - PR Injector

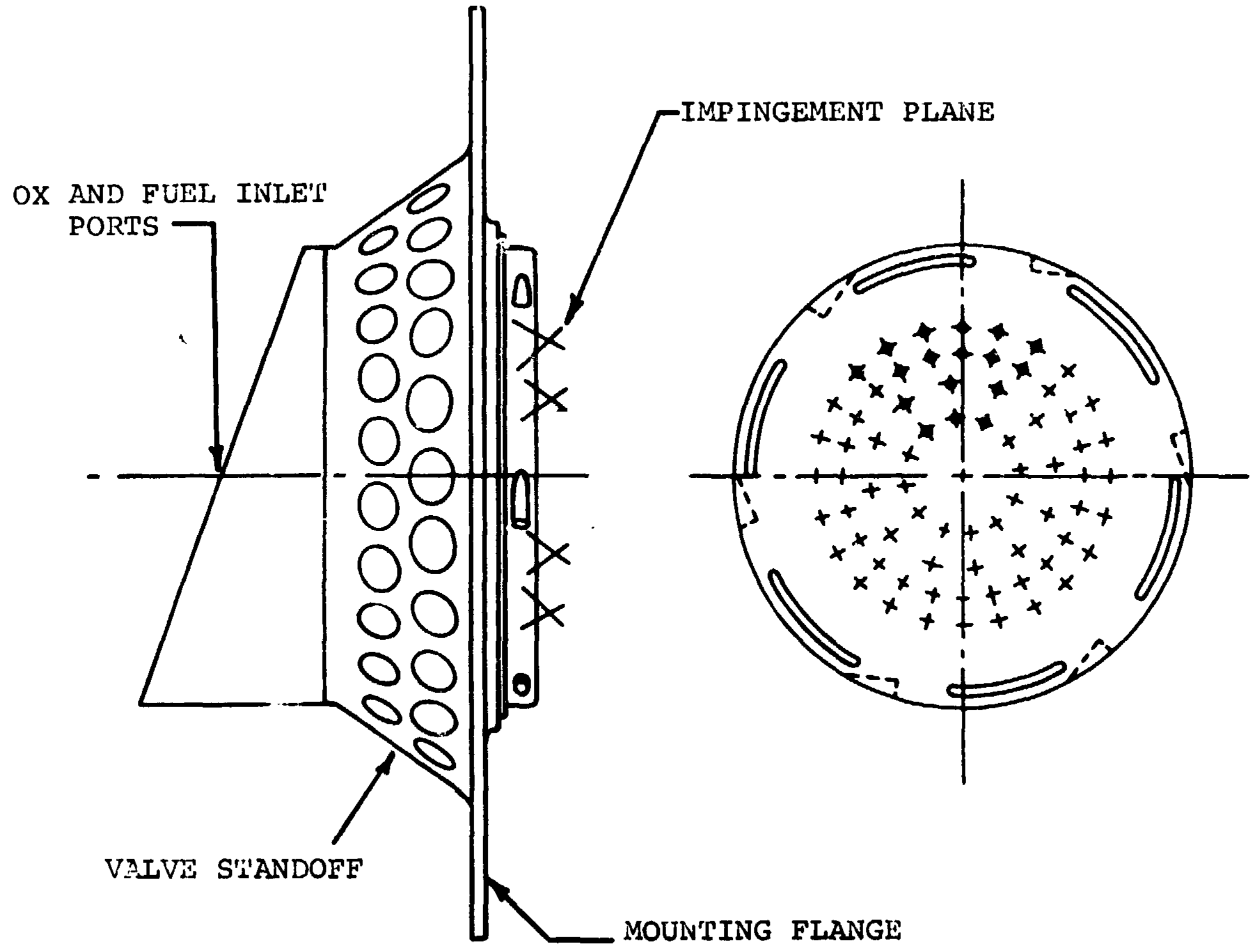


Figure 1-7 Injector SN RDV-DD-PR-1

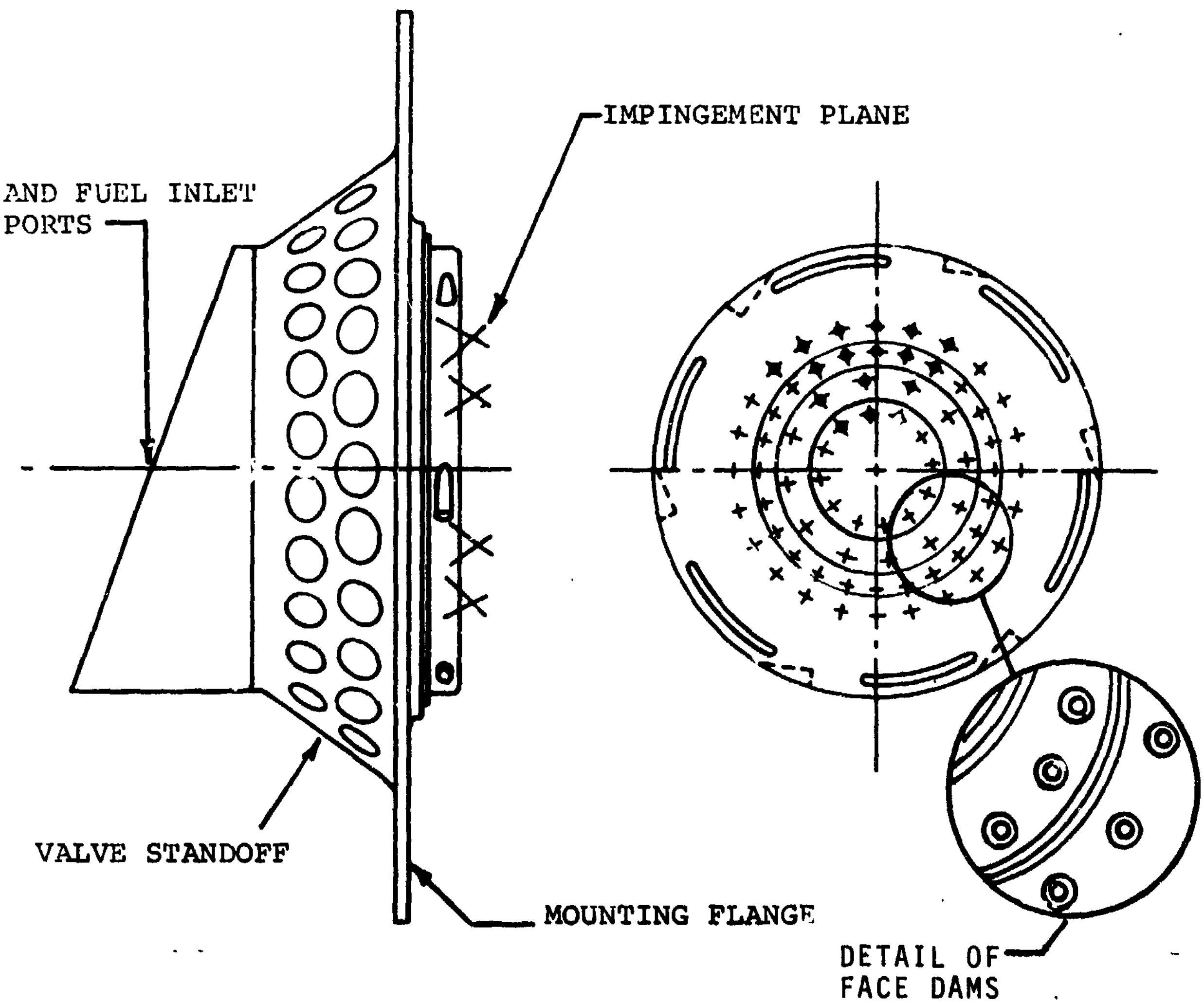


Figure 4.1-8 Injector SN RDV-DD-PR-2

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TORQUE MOTOR BIPROPELLANT VALVE

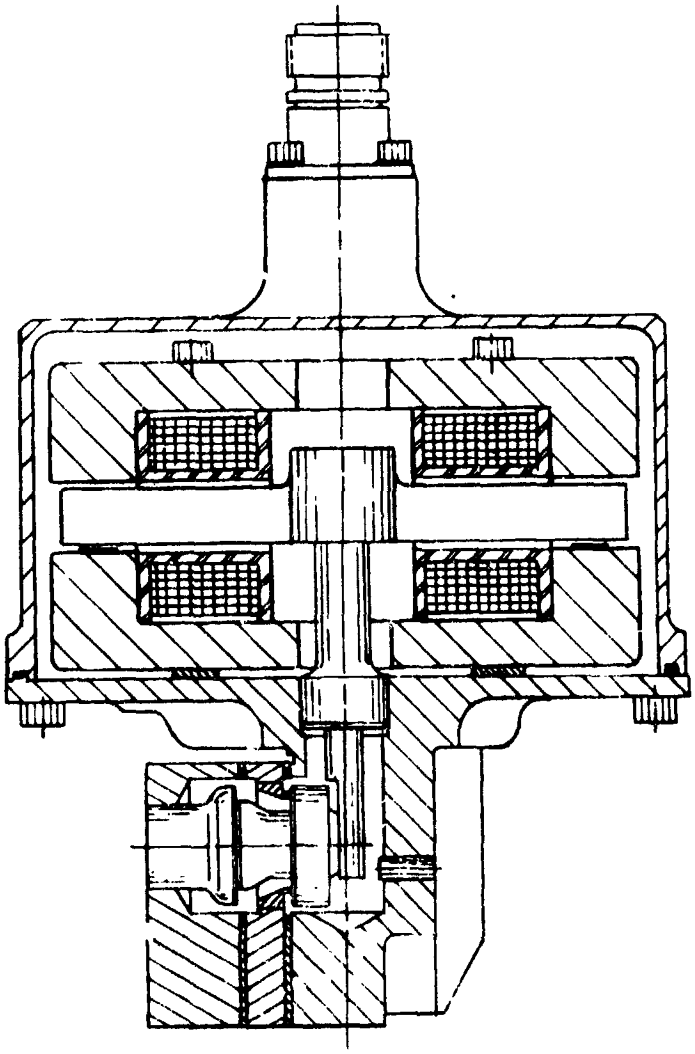


FIGURE 4.1-9

600 LBF TEST VALVE

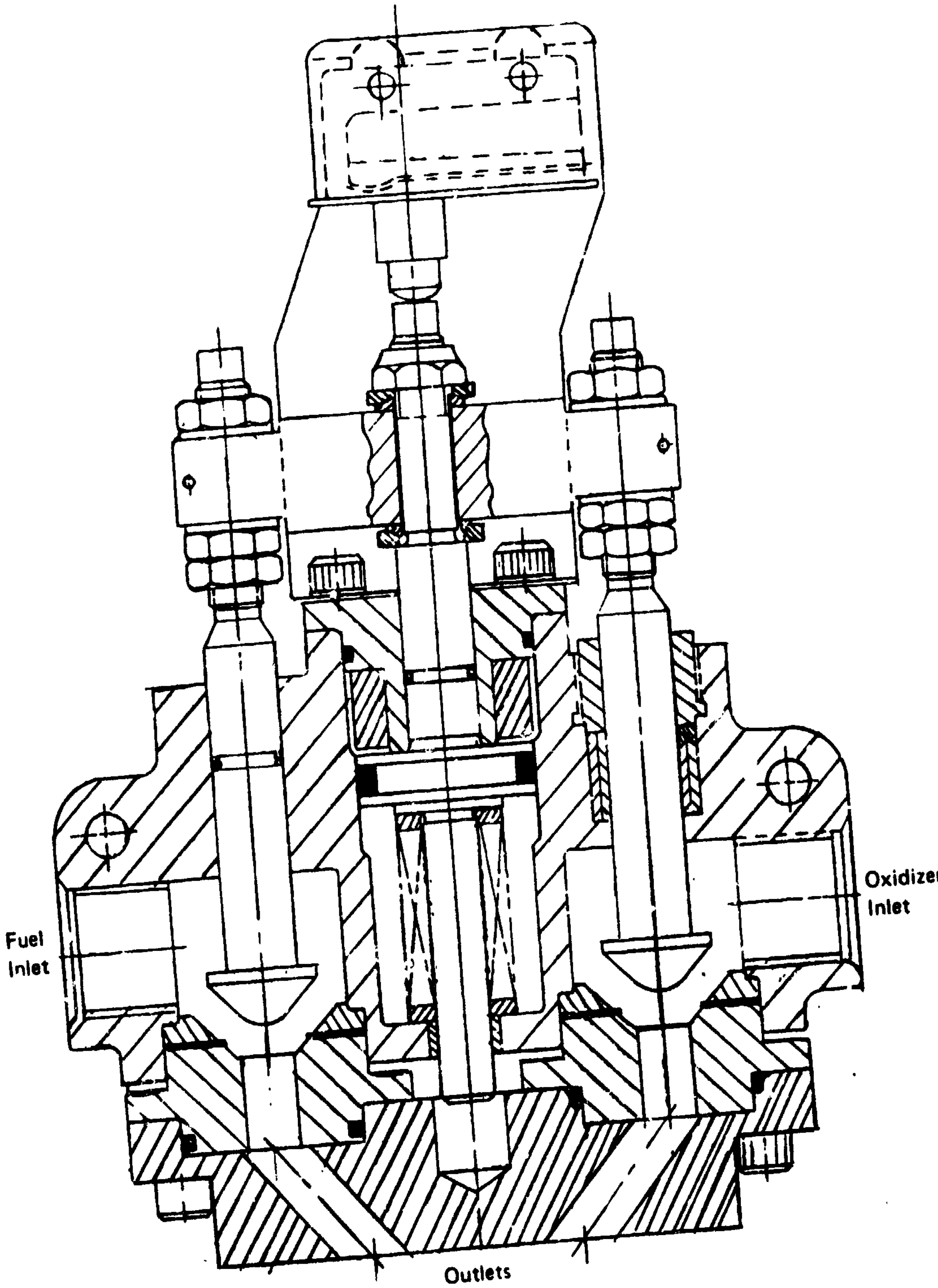


FIGURE 4.1-10

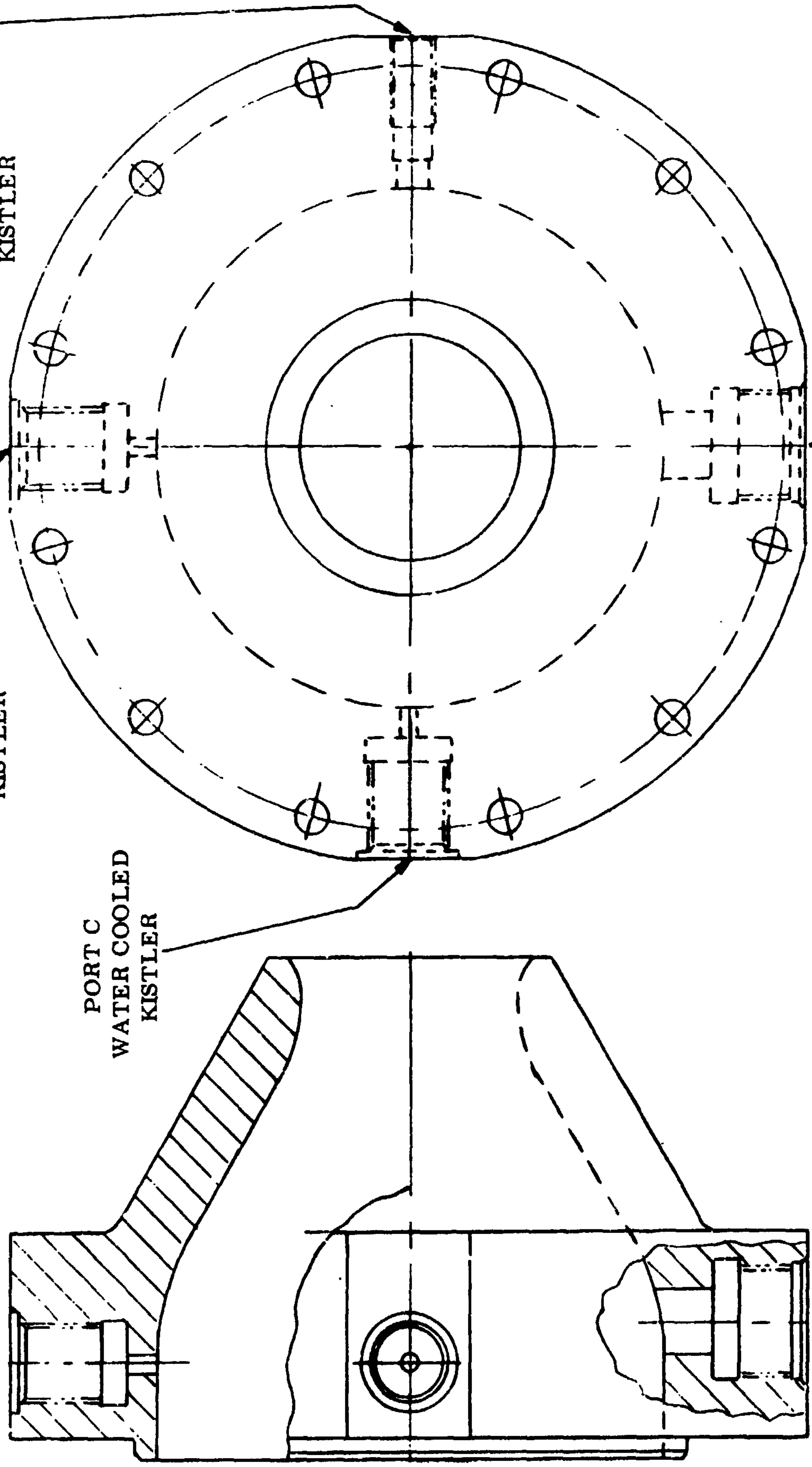


Figure 4.1-11 Bomb Chamber

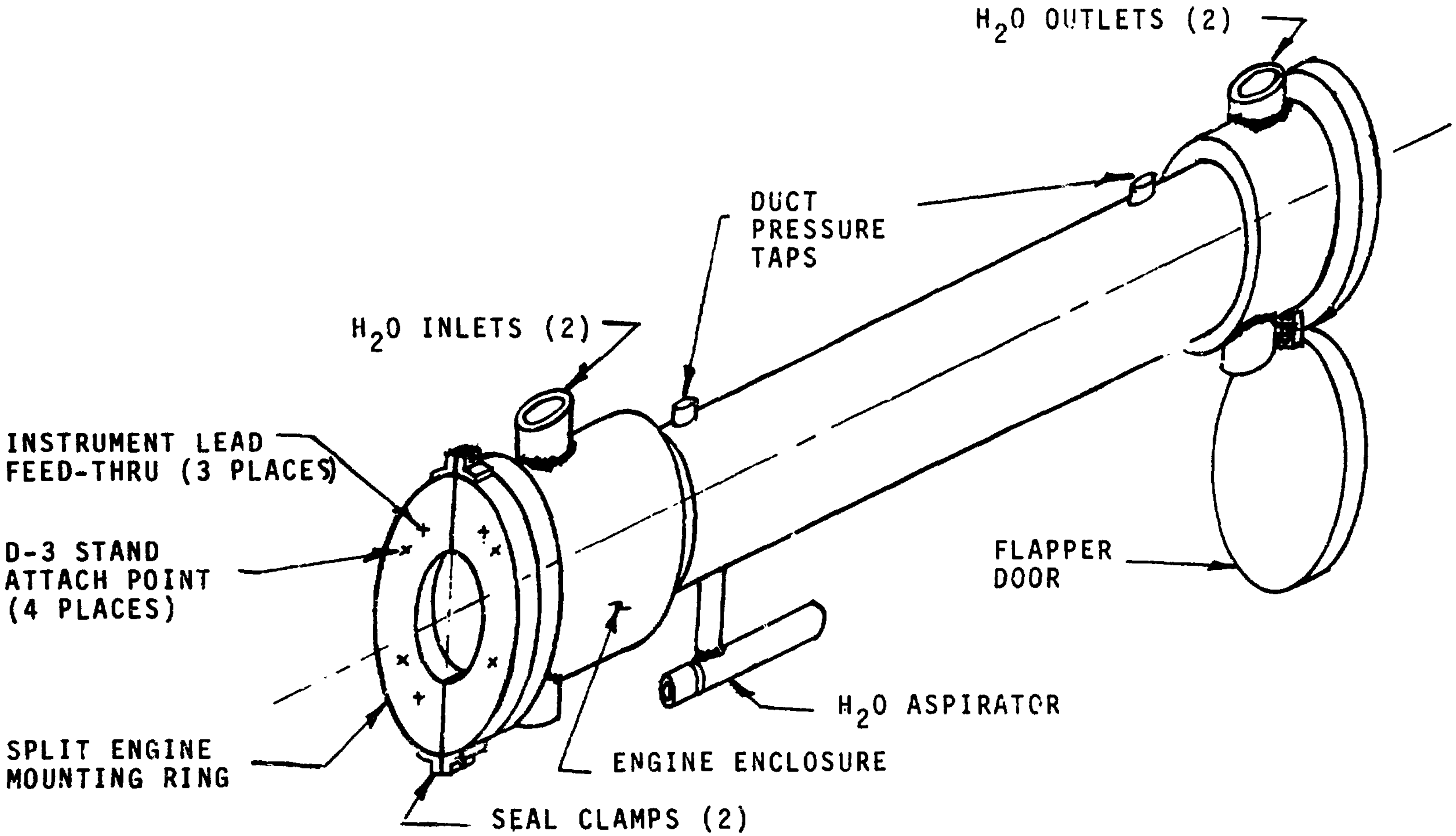


FIGURE 4.1-12. SIMULATED ALTITUDE TEST SET-UP

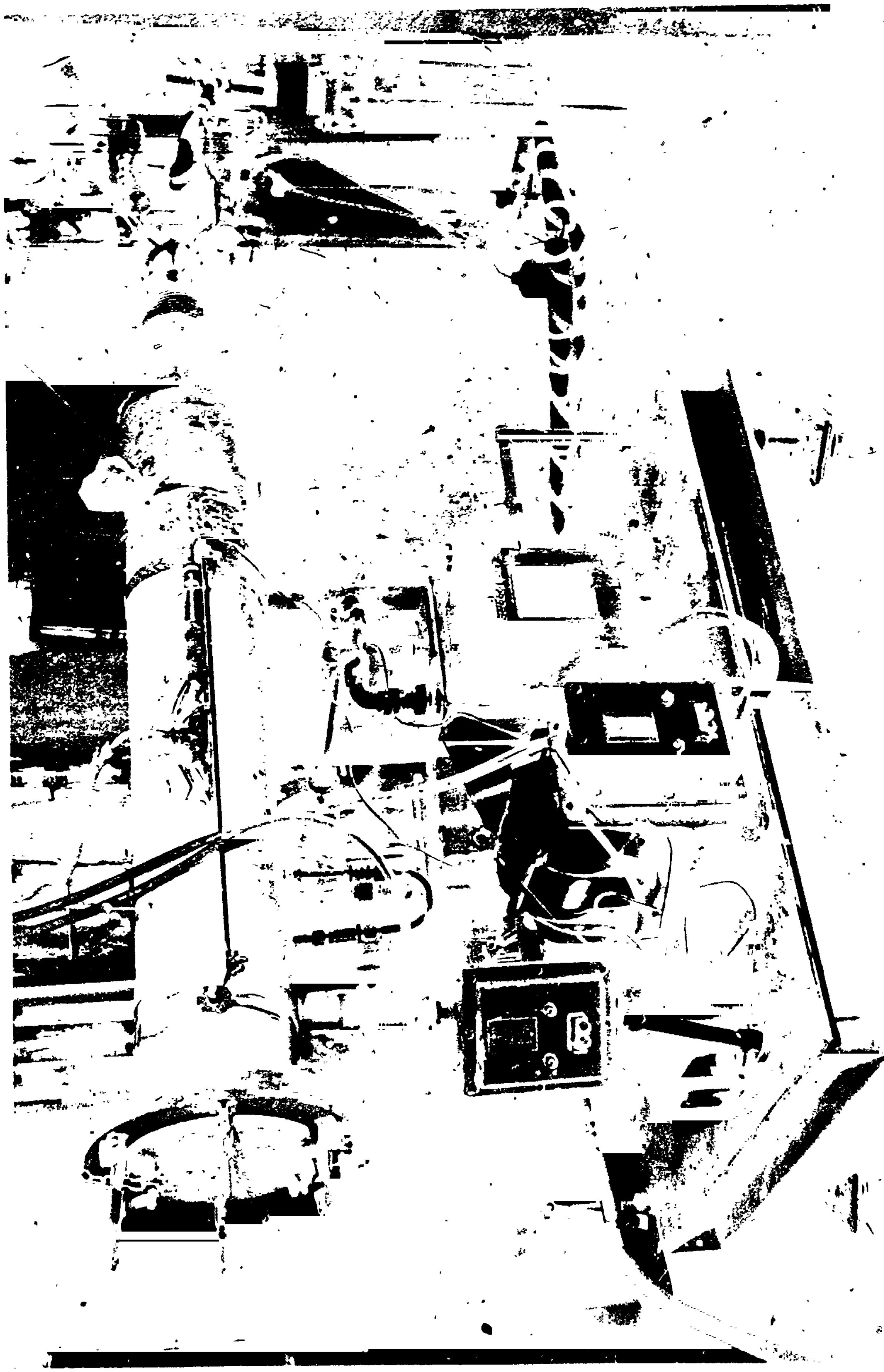


Figure 4.1-13 High Altitude Ignition Test Setup

Figure 4.1-14
Pre B-2-1388
Injector S/N RDV-DD-PR-1
Chamber S/N P-1
Test Valve
ϵ = 31 Extension

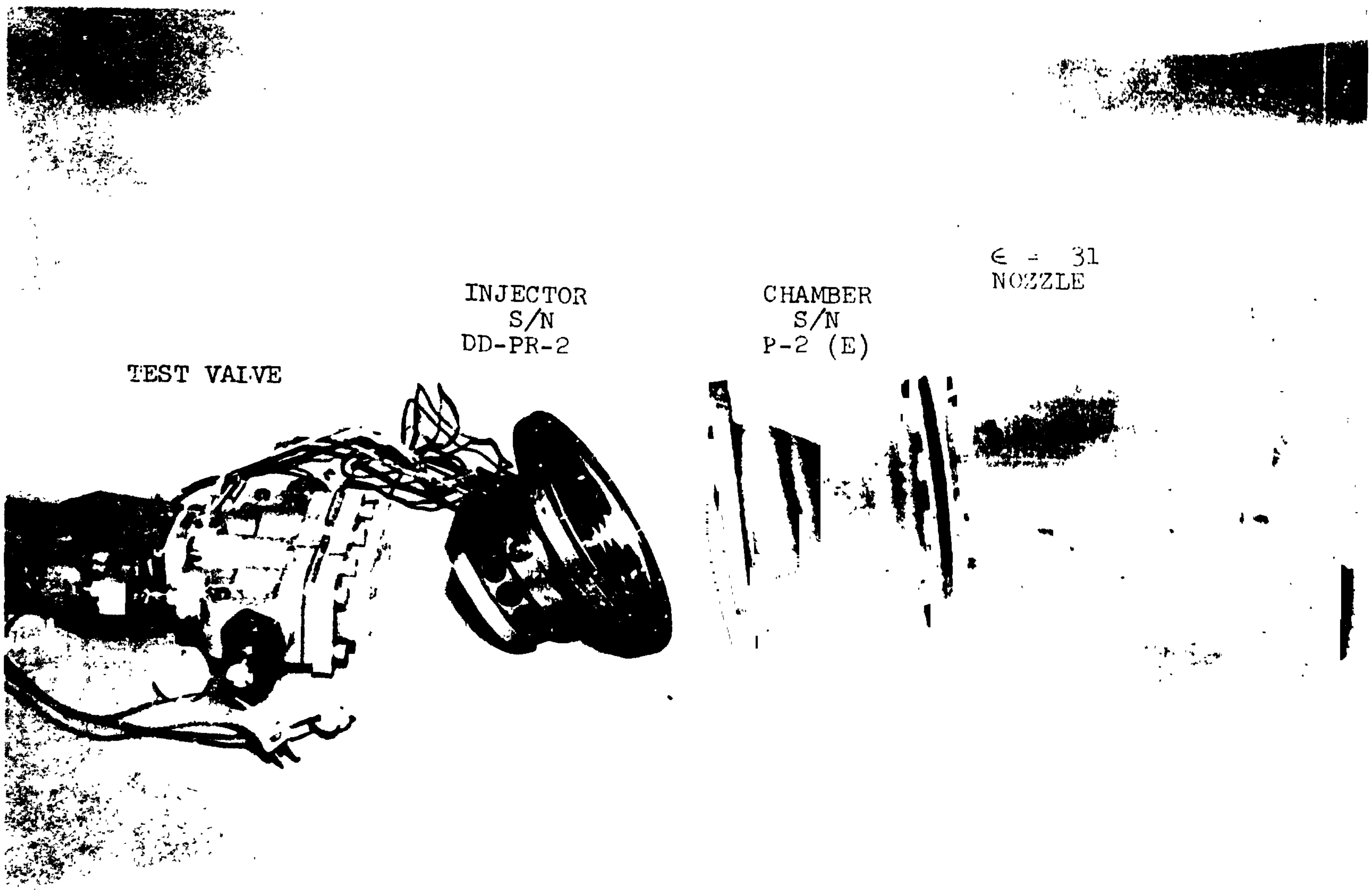


Figure 4.1-15 Post B-2-1512

FOLDOUT

WORST CASE MISSION PERFORMANCE
$\epsilon = 31$

NOTE: THRUST AND VACUUM ISP PROJECTED FROM CHAMBER PRESSURE, FLOW RATES AND THE AVERAGE C_F OBTAINED FROM ALTITUDE TESTING OF THE ENGINE.

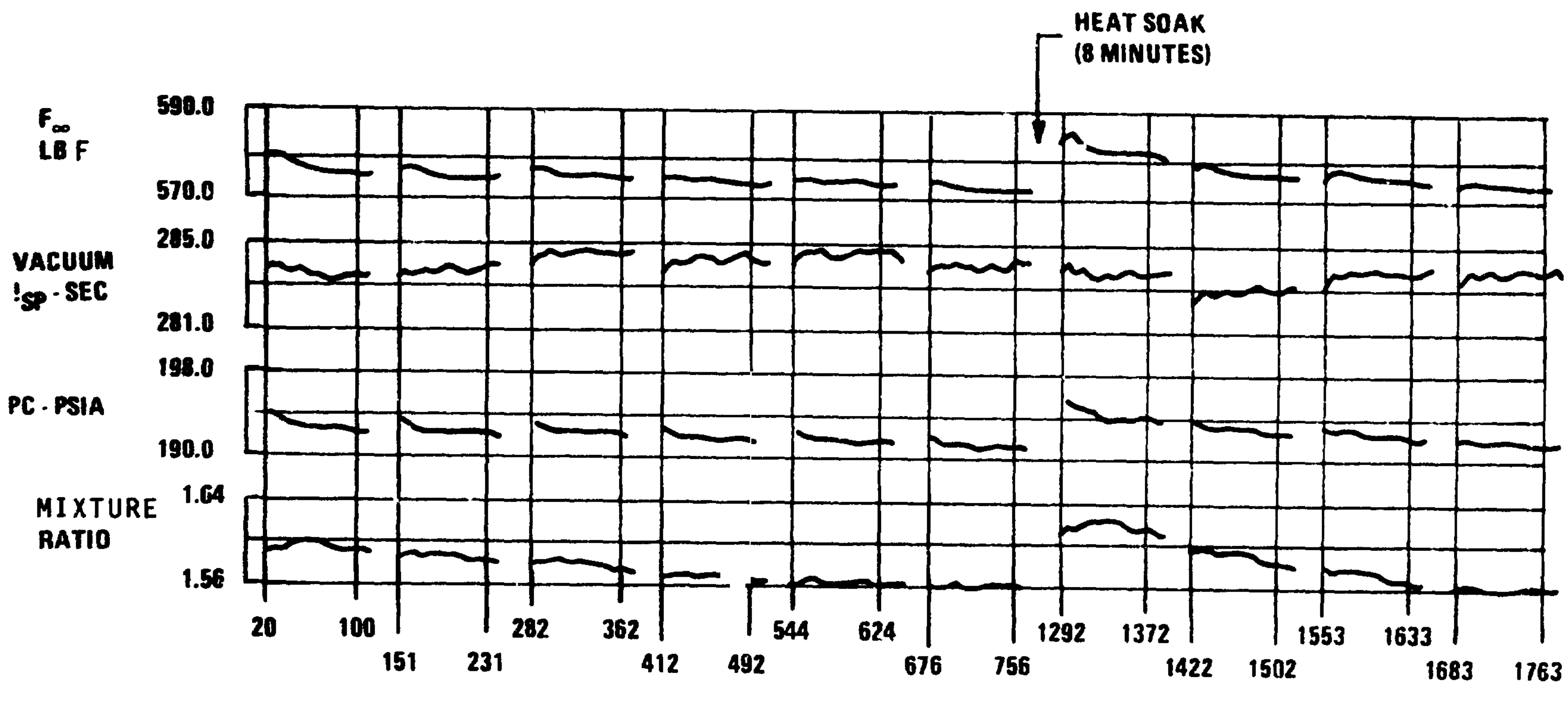
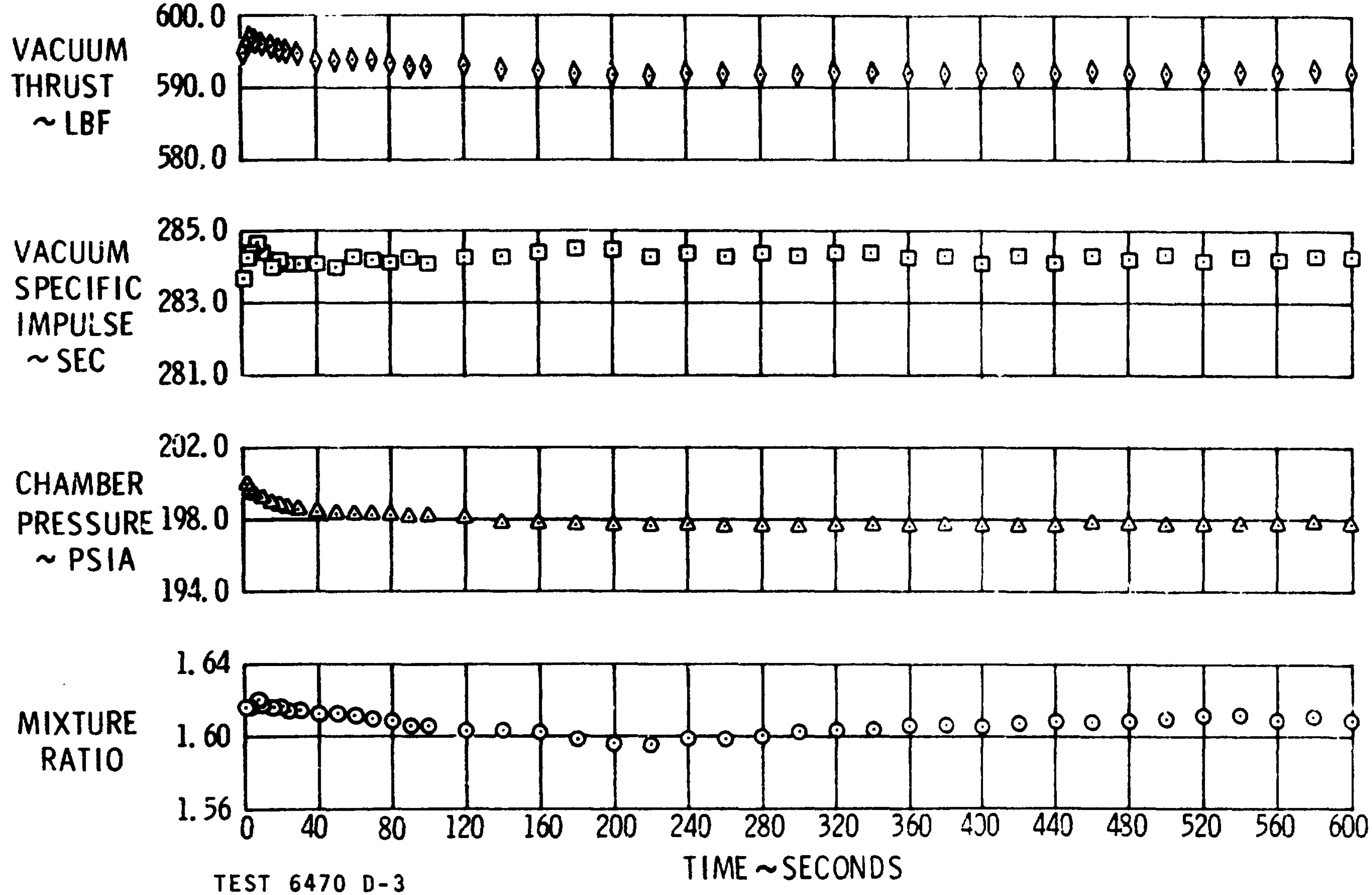


Figure 4.1-16

ENDURANCE TESTING PERFORMANCE - INJECTOR S/N DD-PR-2 - ϵ = 31

NOTE: THRUST AND VACUUM ISP PROJECTED FROM CHAMBER PRESSURE, FLOW RATES AND THE AVERAGE C_F OBTAINED FROM ALTITUDE TESTING OF THE ENGINE.



TEST 6470 D-3

Figure 4.1-17

WORST CASE MISSION PERFORMANCE
HELIUM SATURATED PROPELLANTS
$\epsilon = 31$

NOTE: THRUST AND VACUUM ISP PROJECTED FROM CHAMBER PRESSURE, FLOW RATES AND THE AVERAGE CF OBTAINED FROM ALTITUDE TESTING OF THE ENGINE.

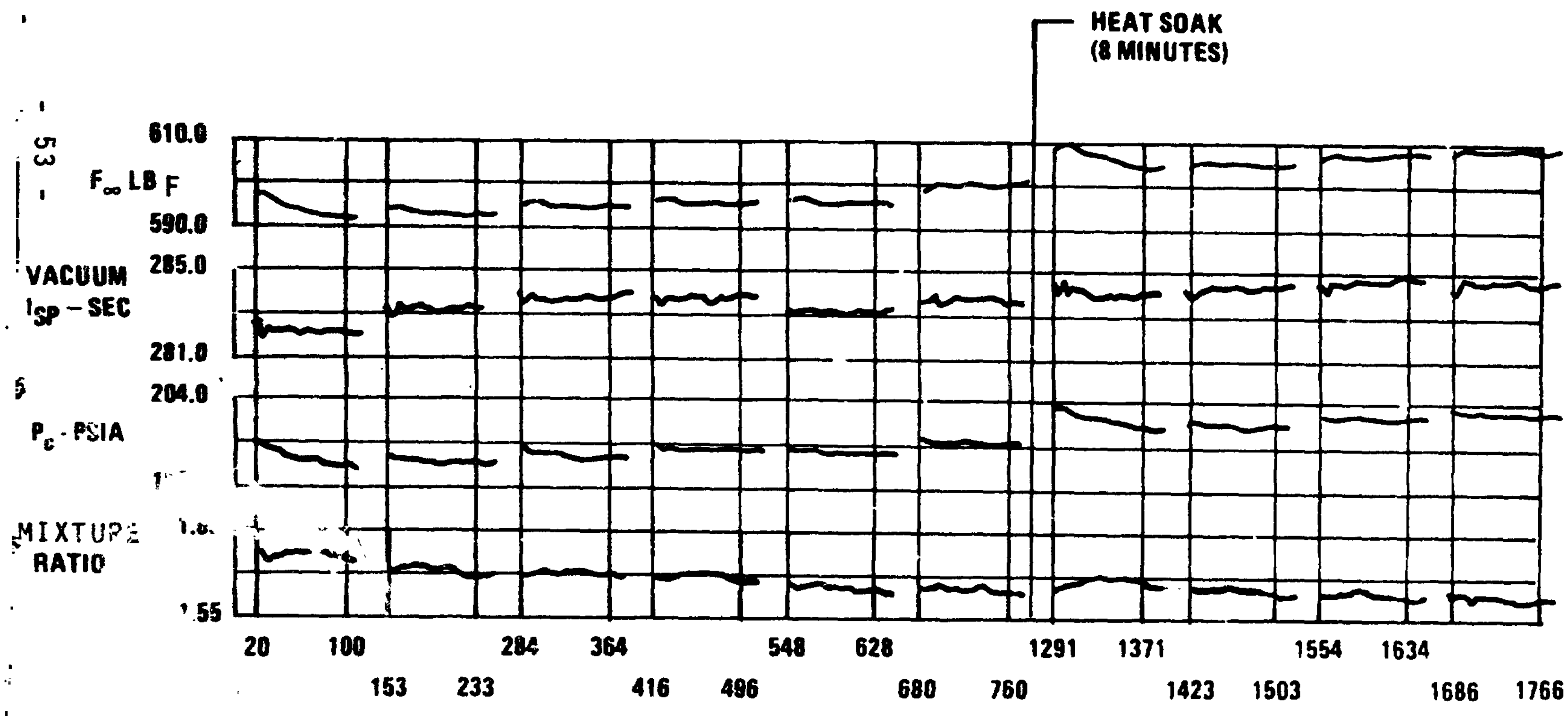


Figure 4.1-18

FOLDOUT

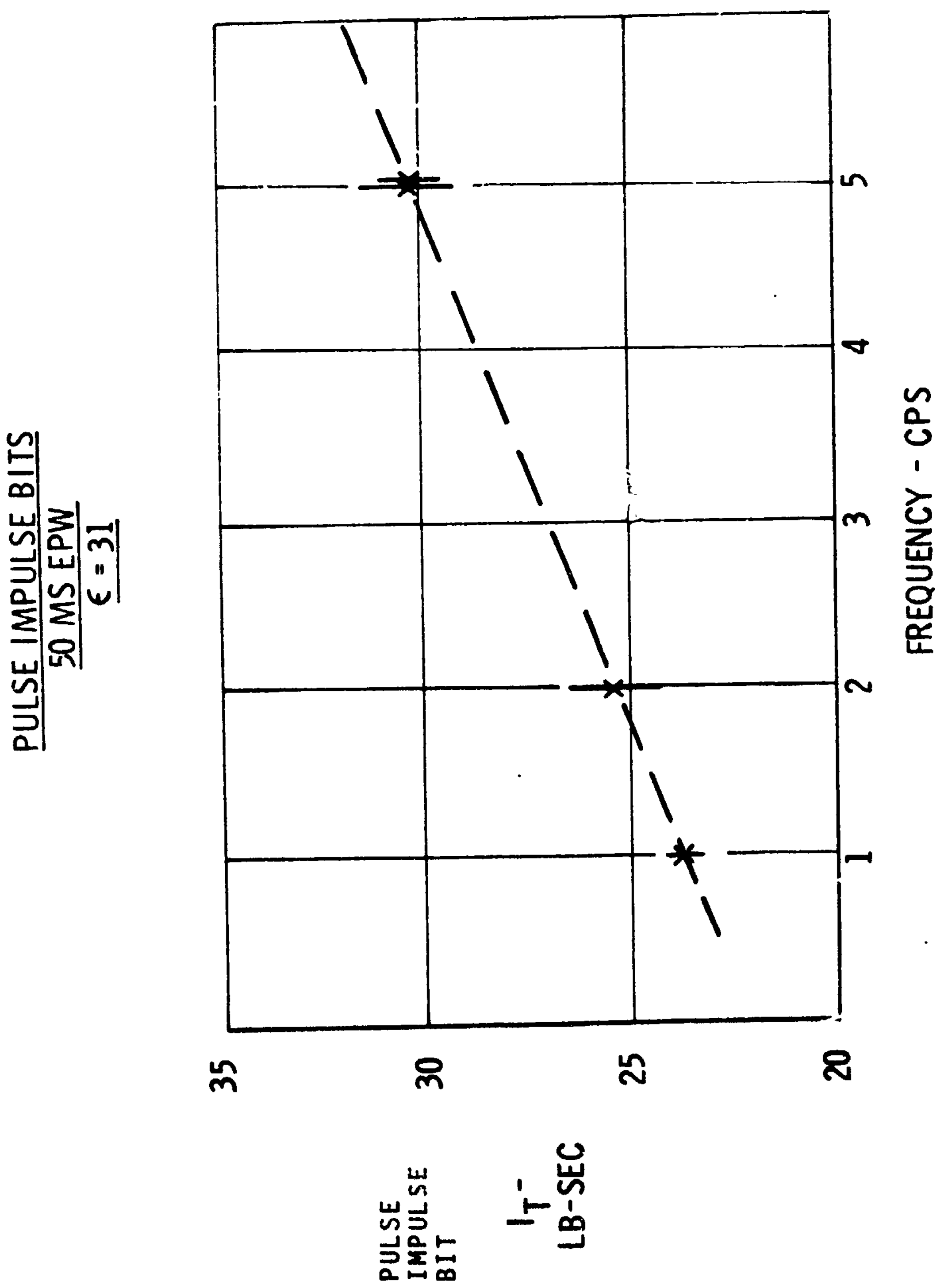


Figure 4.1-19

PULSE IMPULSE BITS COMPARISON

$\epsilon = 31$

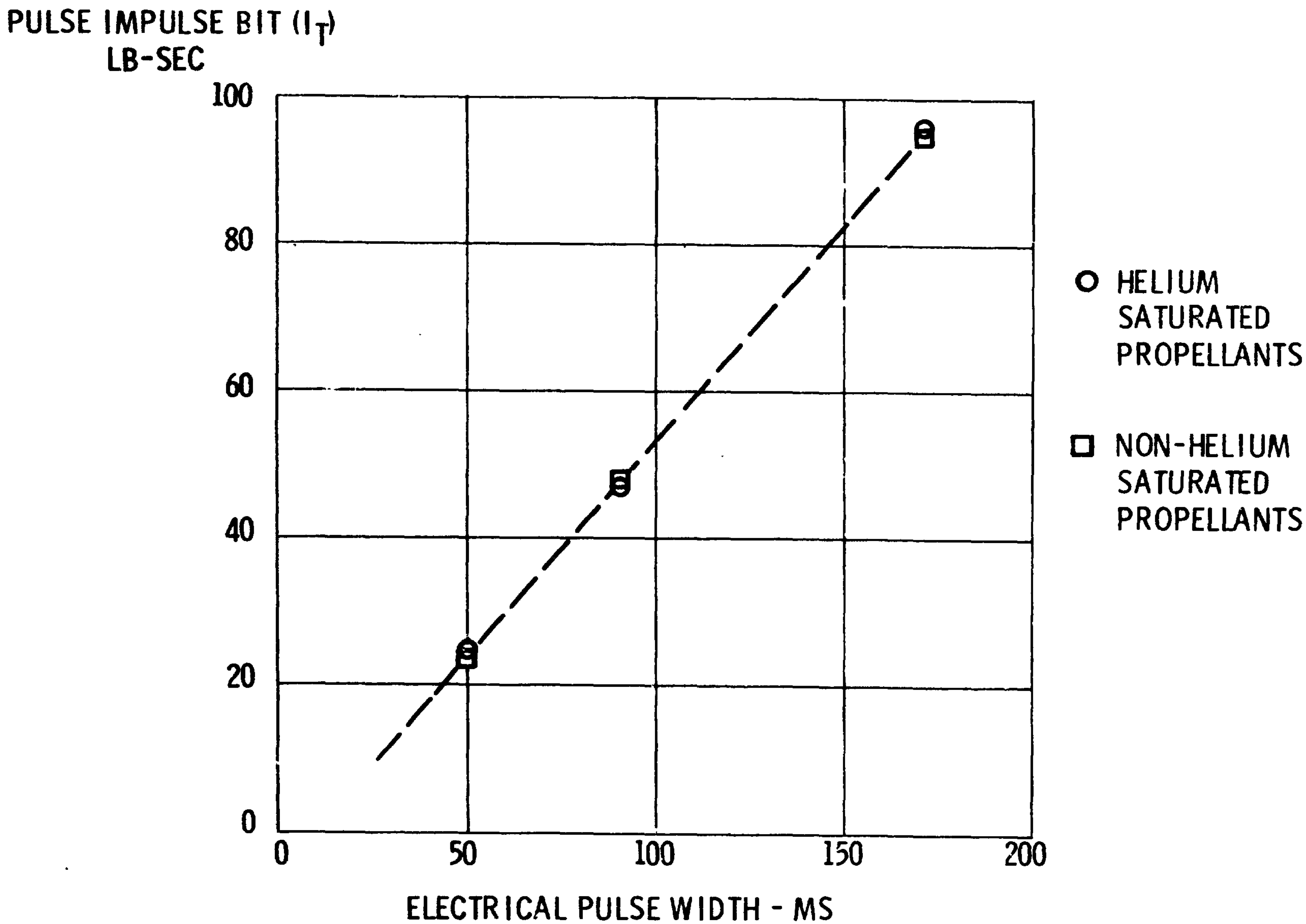


Figure 4.1-20

PULSE $I_{sp\infty}$ VERSUS WIDTH AND FREQUENCY
TESTS B2 - 1411 THROUGH 1422

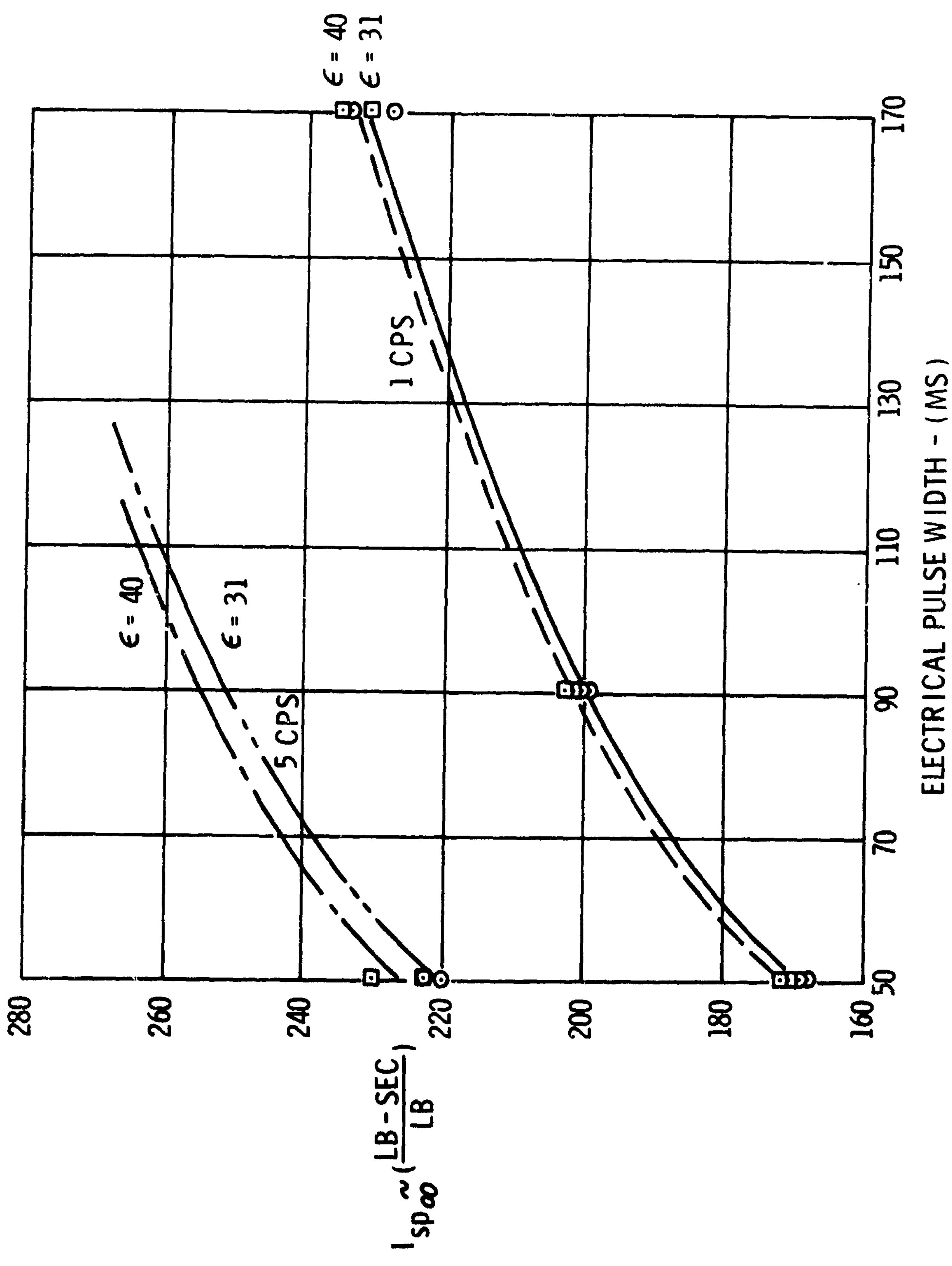


Figure 4.1-21

PREDICTED FLIGHT ENGINE PULSE I_{sp} VERSUS WIDTH AND FREQUENCY

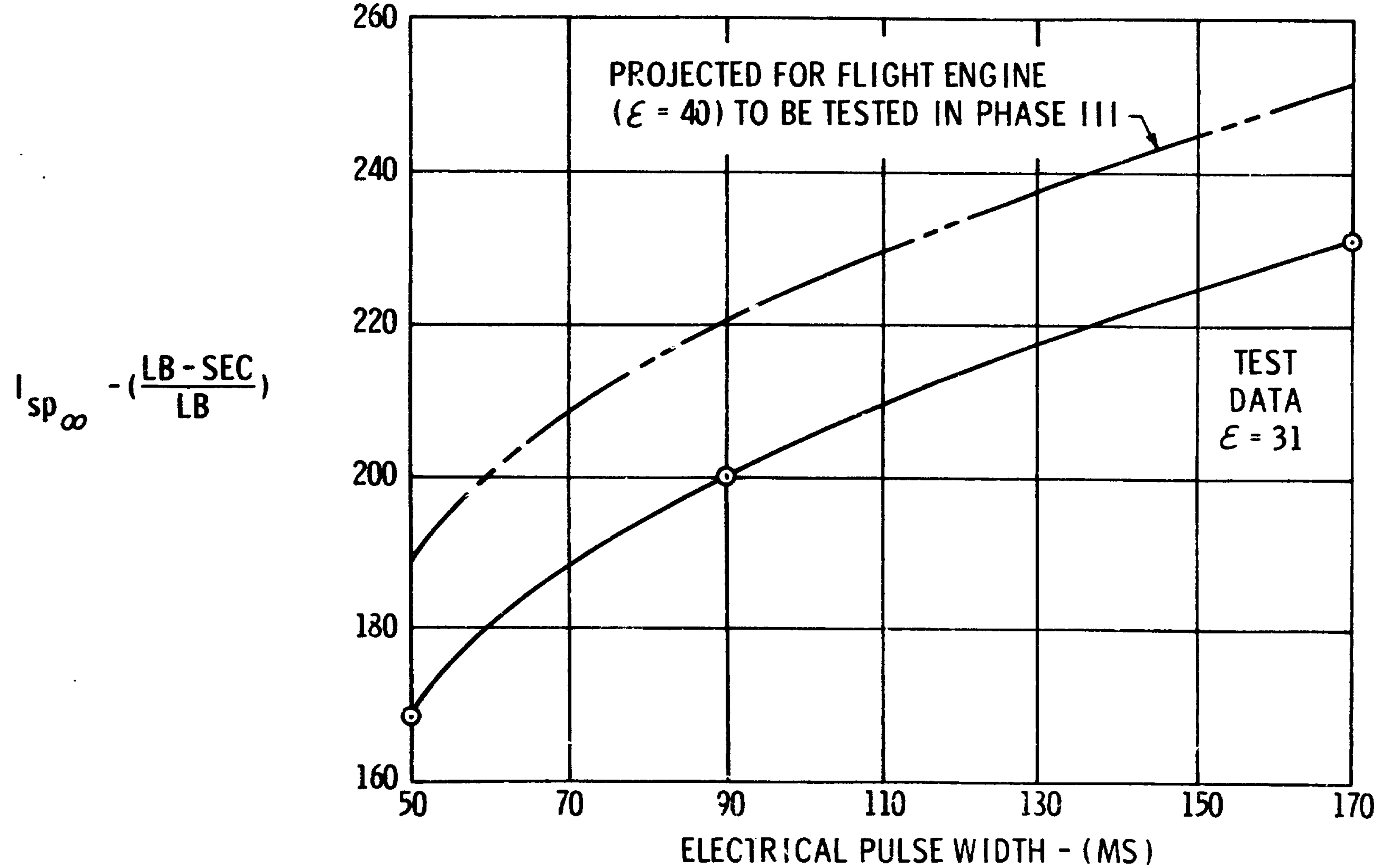


Figure 4.1-22

I_{SP} - CSTAR AND T_{MAX} VERSUS BARRIER RATIO - ρ

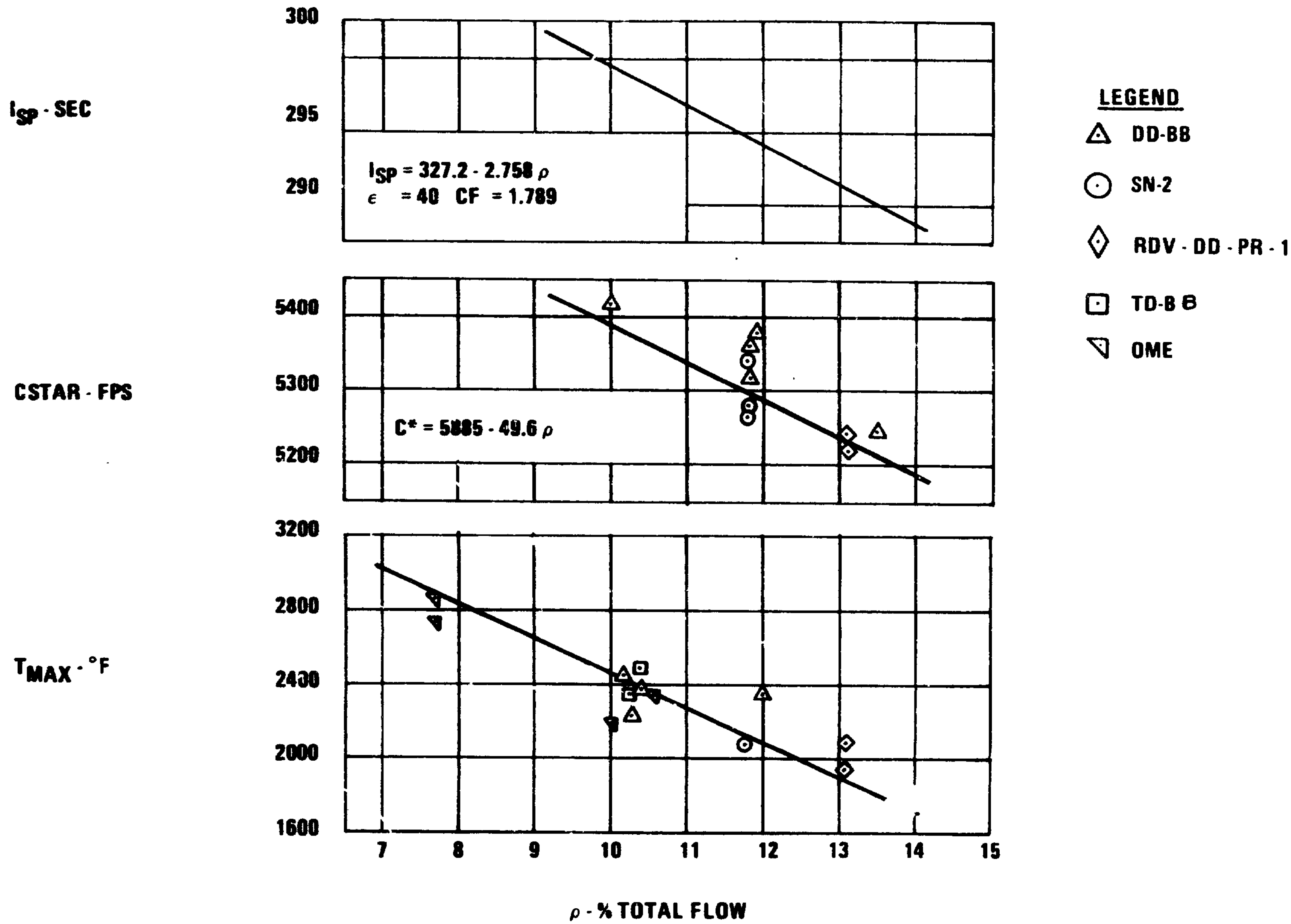


FIGURE 4.1-23

SCALING FACTOR
T_{MAX} VERSUS ρ^* - N_2O_4/MMH
INSULATED ENGINE

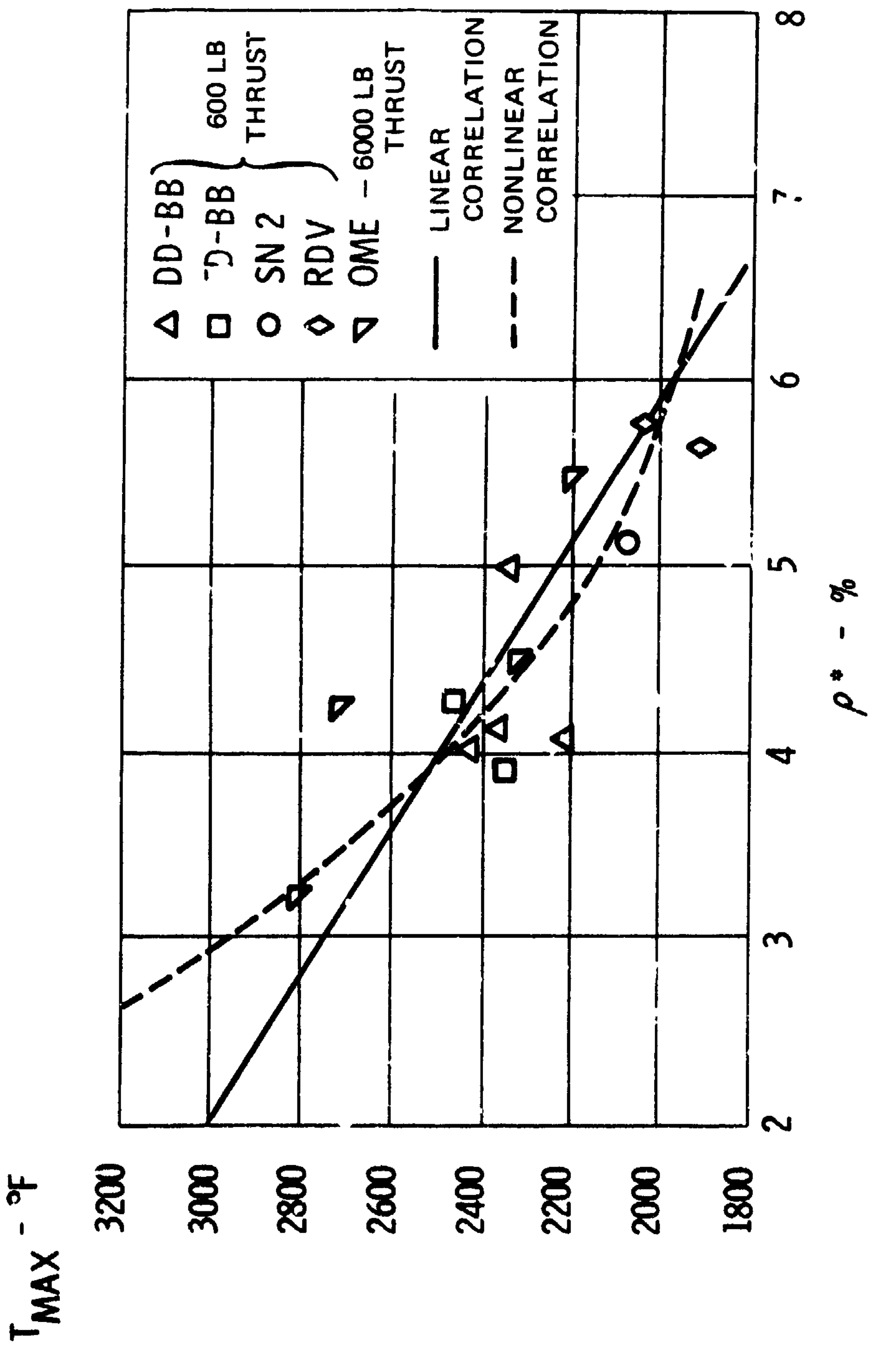
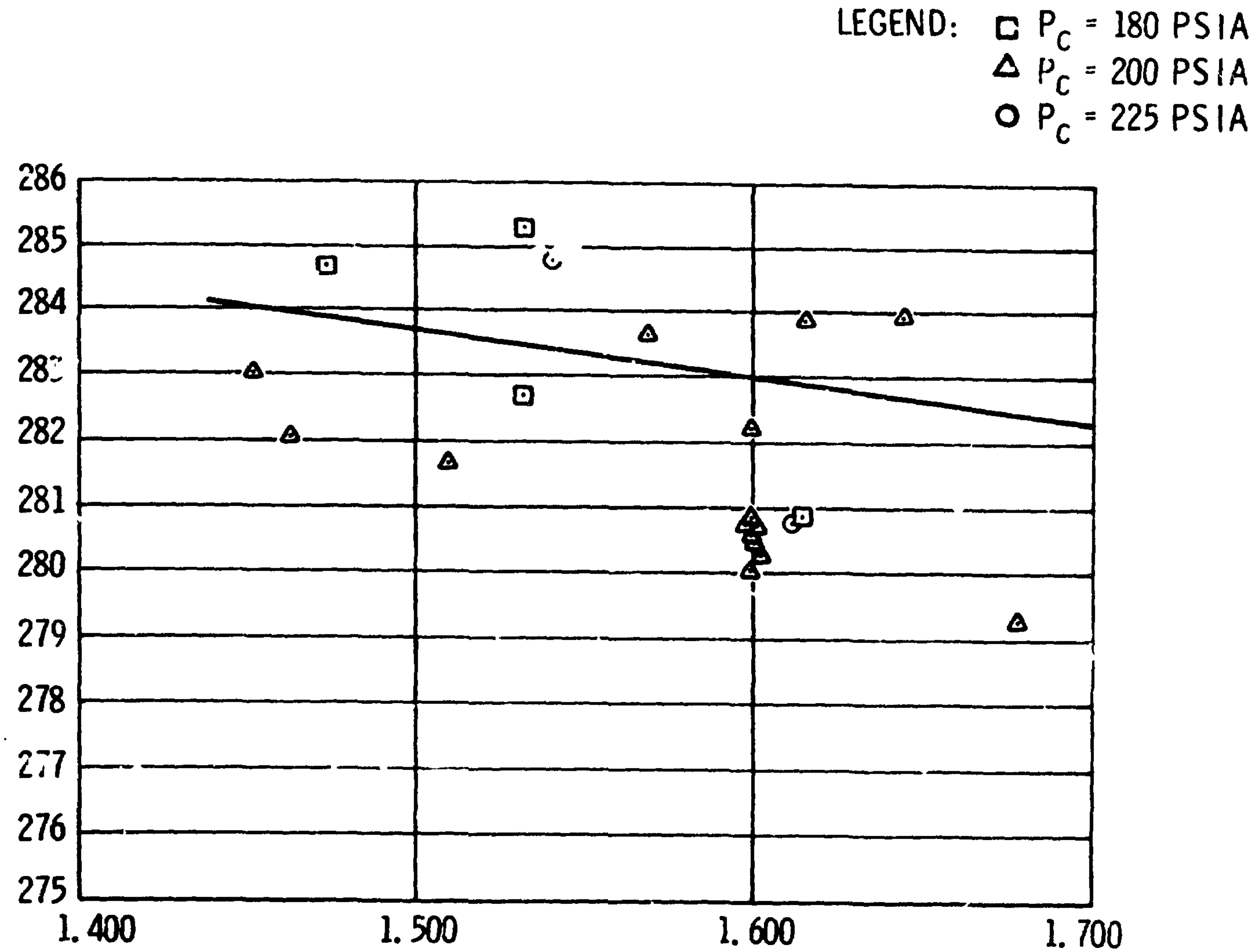


FIGURE 4.1-24

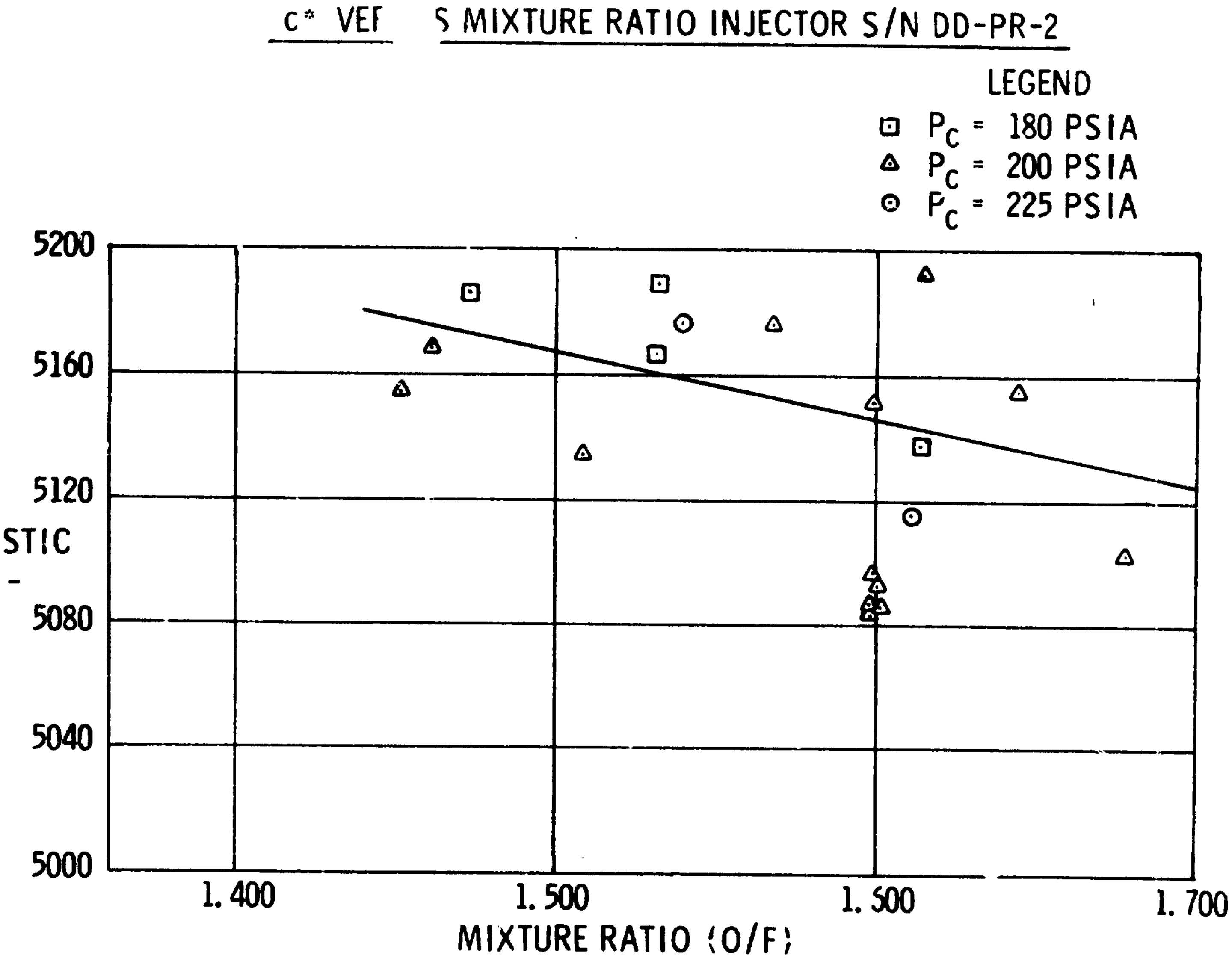
VACUUM SPECIFIC IMPULSE VERSUS MIXTURE RATIO

INJECTOR $\frac{\text{S/N DD-PR-2}}{\varepsilon = 31}$



MIXTURE RATIO (O/F)

FIGURE 4.1-25


c* VEI S MIXTURE RATIO INJECTOR S/N DD-PR-2
LEGEND
P_C = 180 PSIA
P_C = 200 PSIA
P_C = 225 PSIA
CHARACTERISTIC VELOCITY - FT/SEC
5200
5160
5120
5080
5040
5000
1.400
1.500
1.500
1.700
MIXTURE RATIO (O/F)

FOGIRE 4.1-26

c* VERSUS MIXTURE RATIO INJECTOR
S/N RDV-DD-PR-1

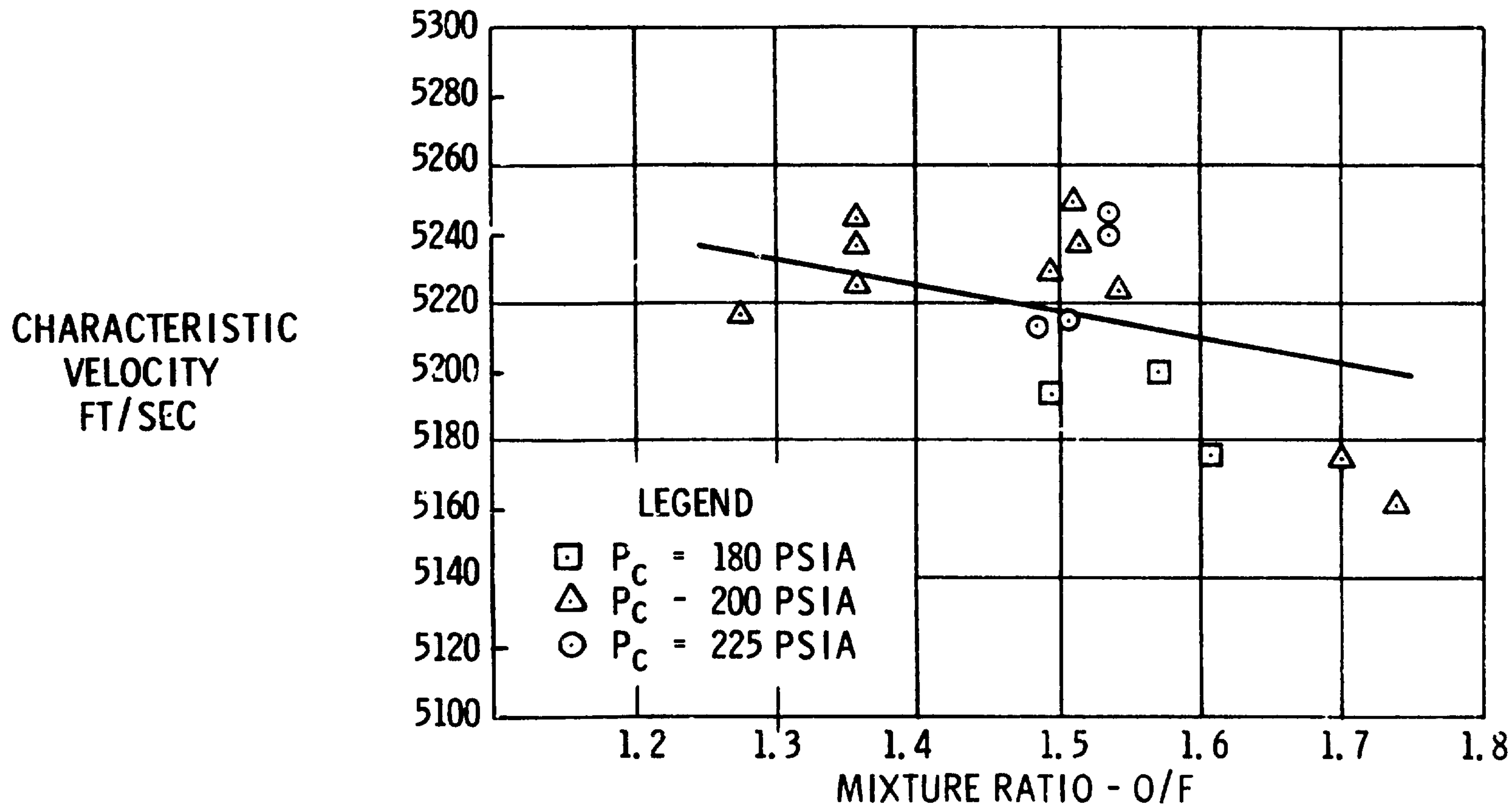


FIGURE 4.1-27

VARIATION OF C* WITH OPERATING CONDITION

ρ = 12.0

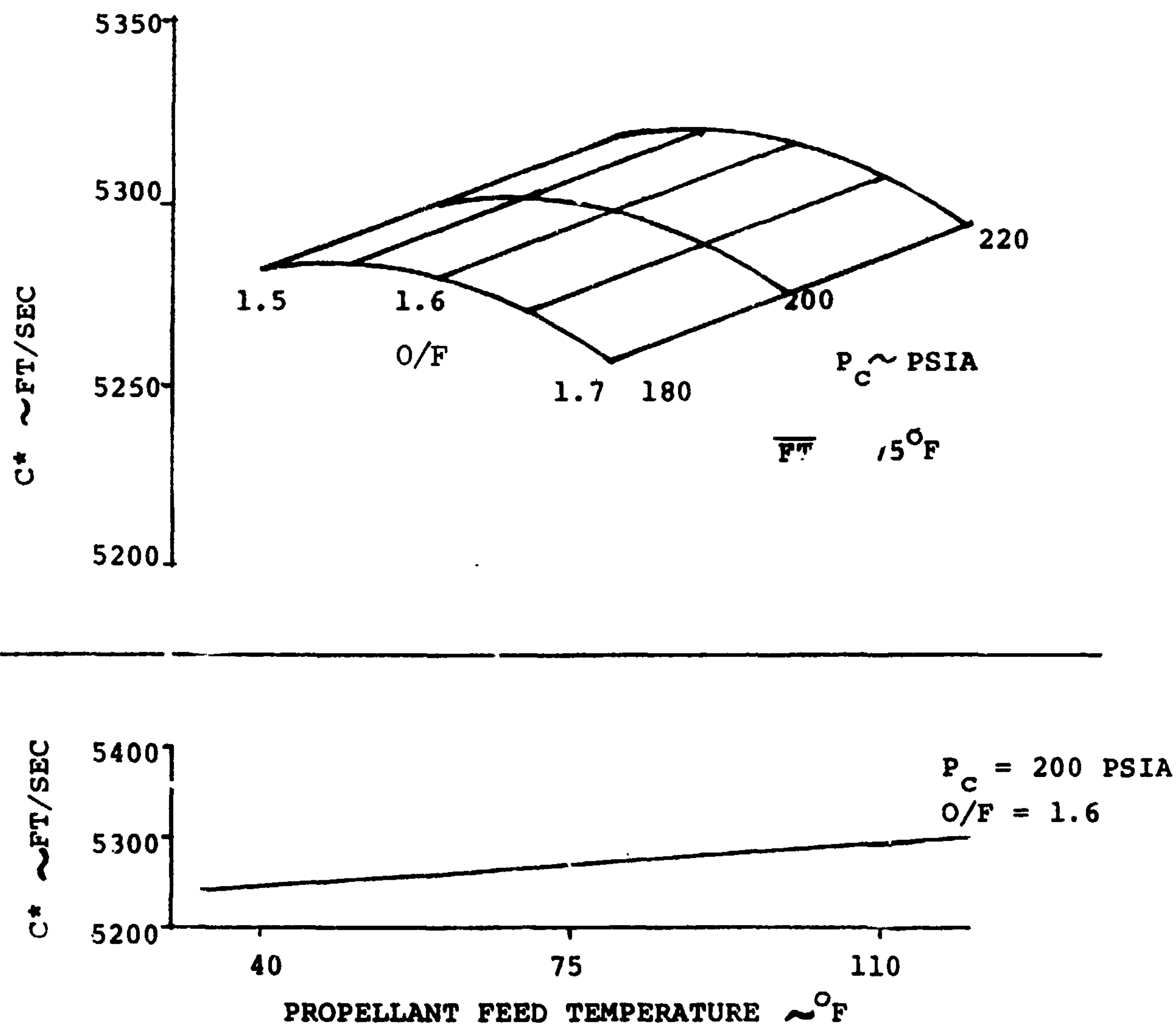
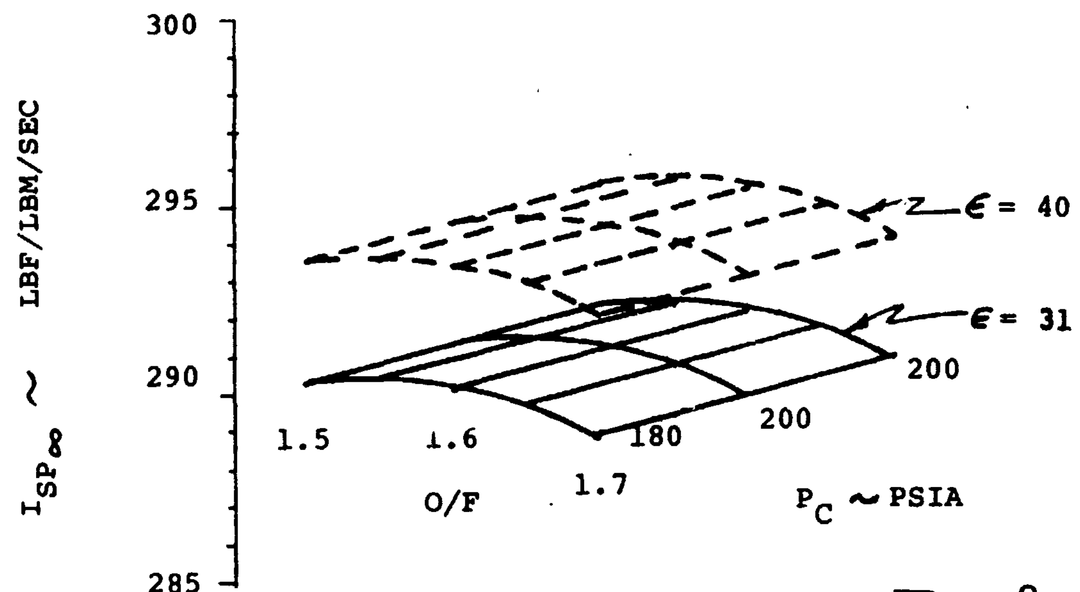


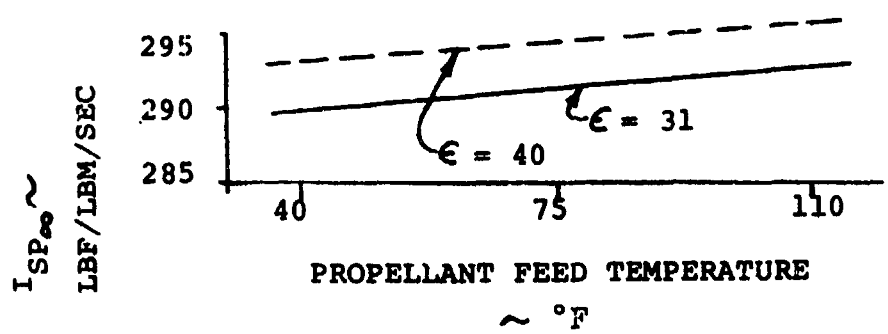
FIGURE 4.1-28

VARIATION OF I_{SP} WITH OPERATING CONDITION

ρ = 12.0 %



$\overline{FT}$ = 75°F



P_C = 200 PSIA
O/F = 1.6

FIGURE 4.1-29

VARIATION OF MAXIMUM THROAT TEMPERATURE WITH OPERATING CONDITION

$\rho = 12.0$

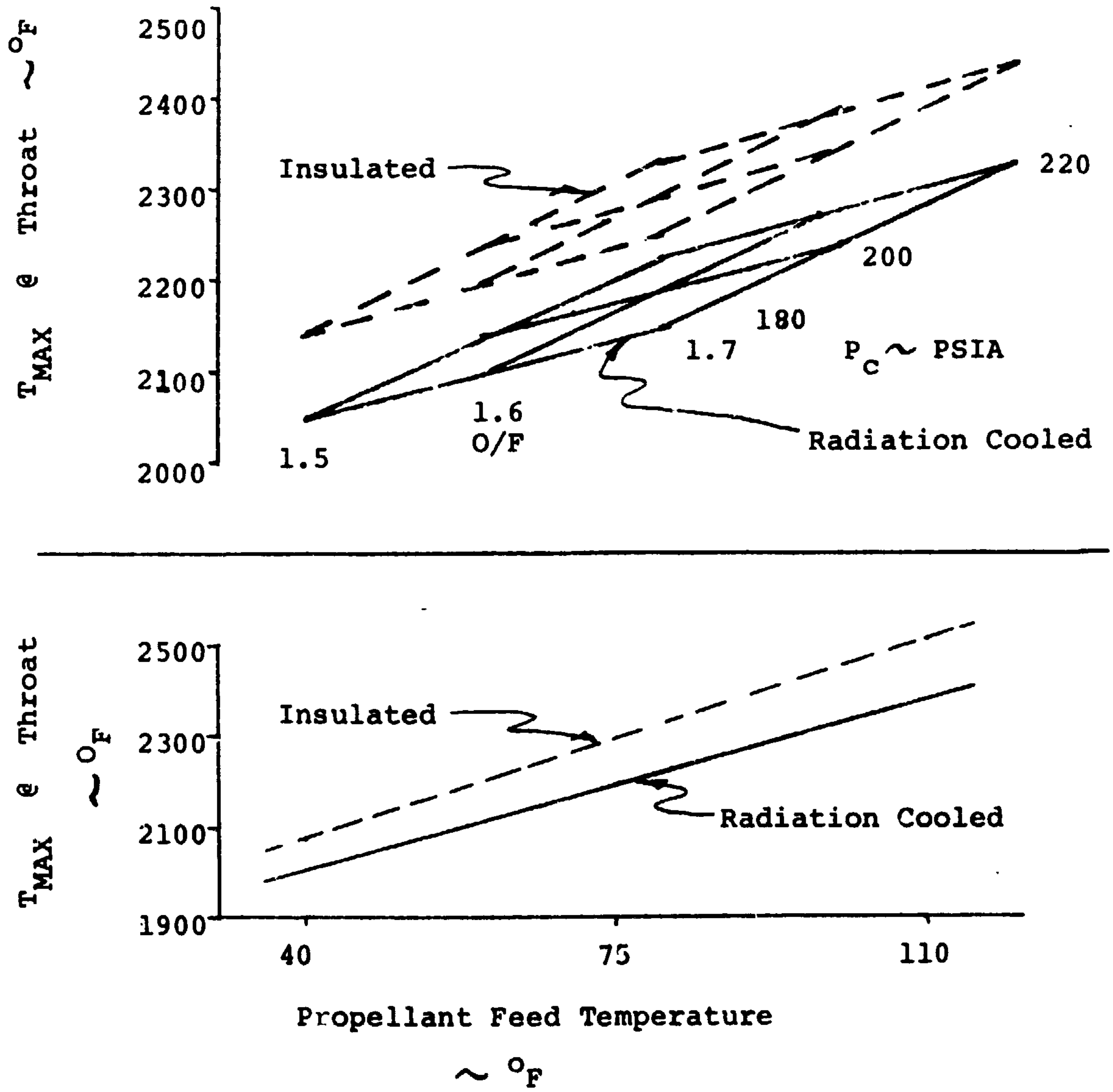


FIGURE 4.1-30

PULSING PERFORMANCE
VS.
STEADY STATE OPERATING CONDITION

EPW = 50 MS

f = 1 CPS

PROPELLANT TEMPERATURES = 75°F

INJECTOR S/N DD-PR-2 (ε = 31)

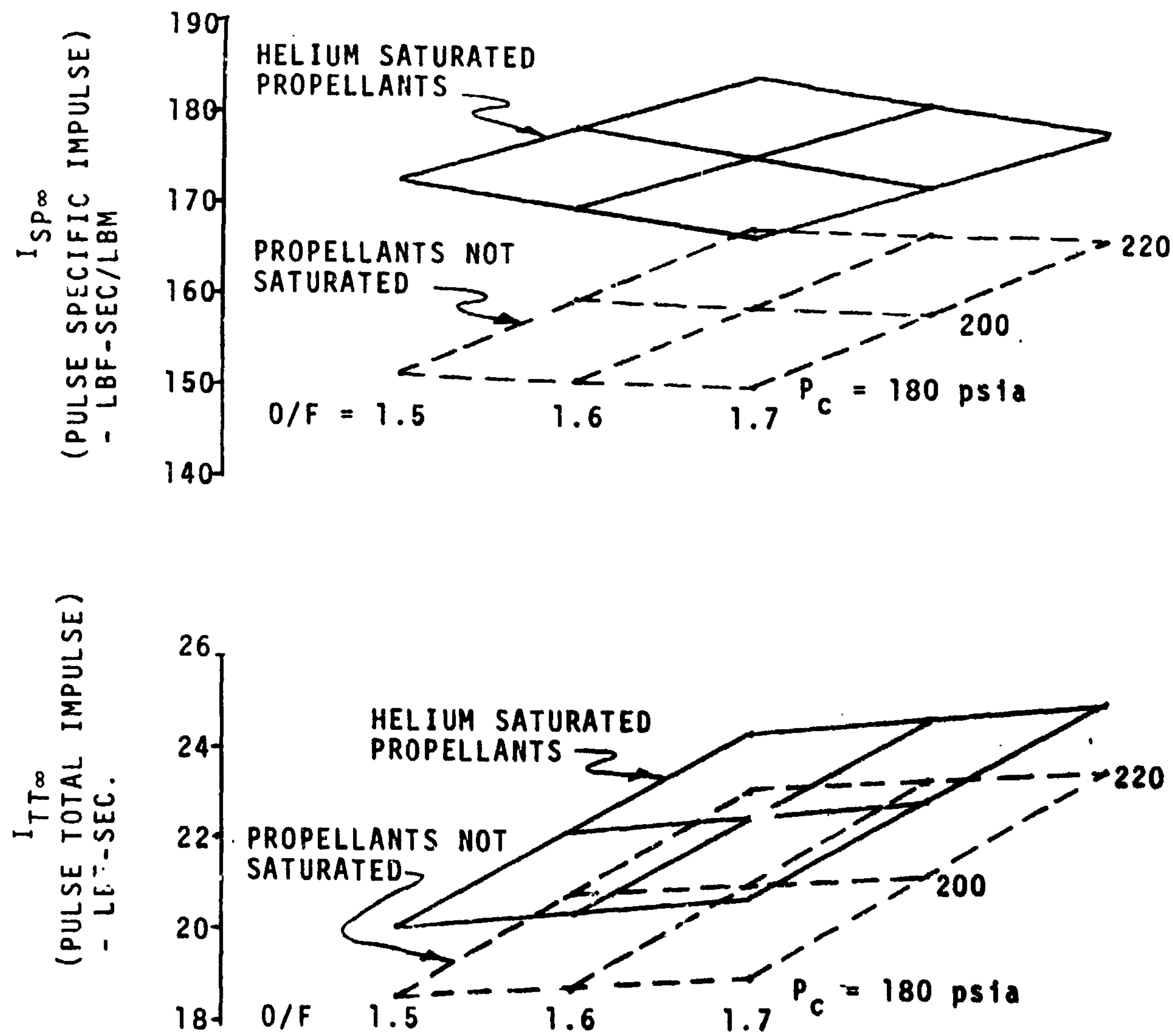
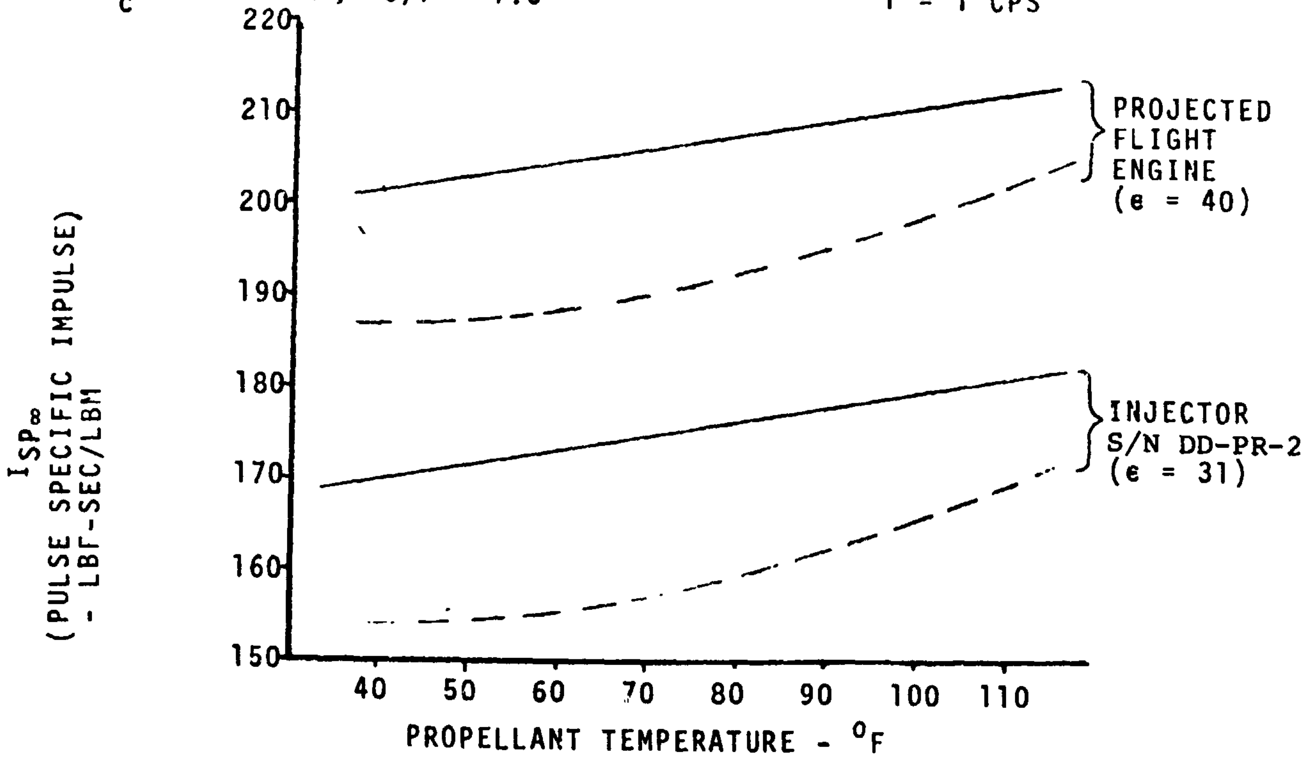


FIGURE 4.1-31

PULSING PERFORMANCE
VS.
PROPELLANT TEMPERATURE

STEADY STATE OPERATING CONDITION: EPW = 50 MS

P_c = 200 PSIA, O/F = 1.6 f = 1 CPS



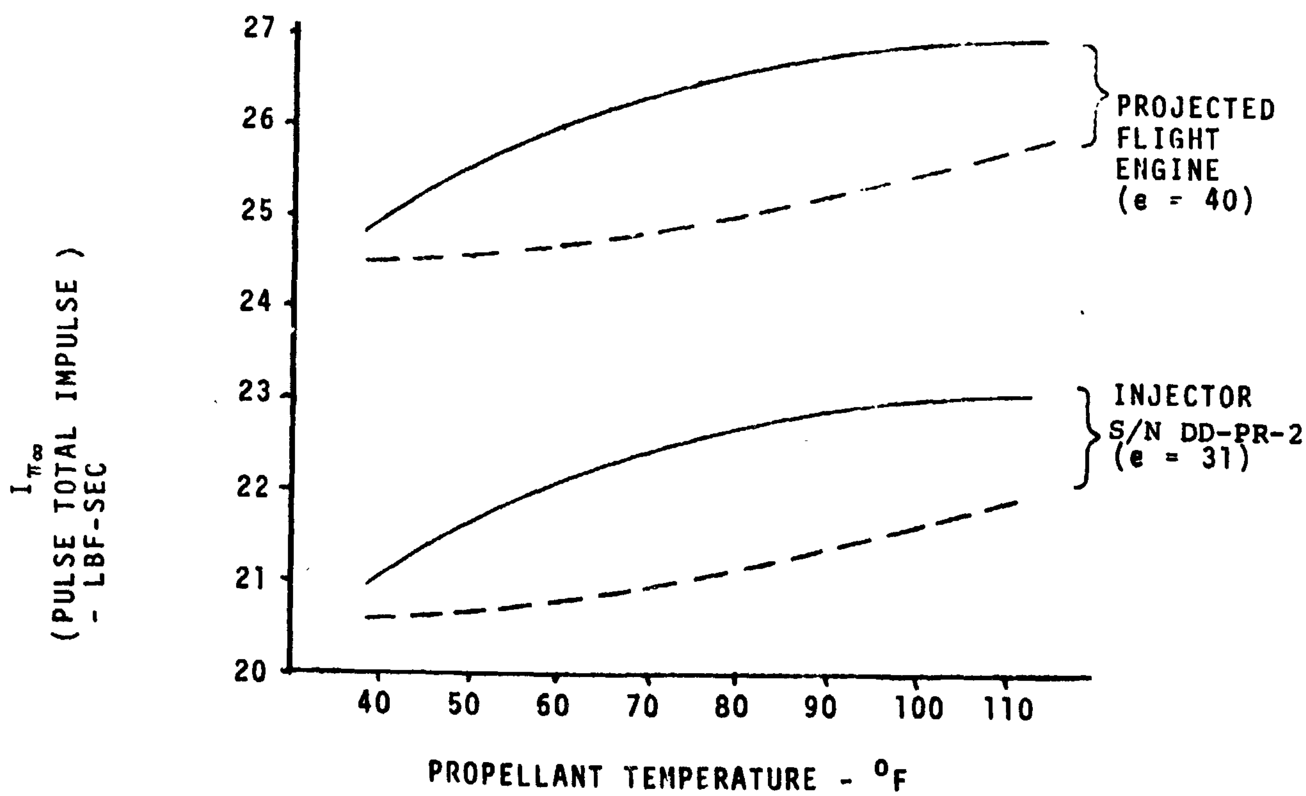


FIGURE 4.1-32

PULSING PERFORMANCE

VS.

ELECTRICAL PULSE WIDTH

STEADY STATE OPERATING CONDITION:

P_c = 200 PSIA

O/F = 1.6

PROPELLANT TEMPS. = 75°F

f = 1 CPS

INJECTOR S/N DD-PR-2

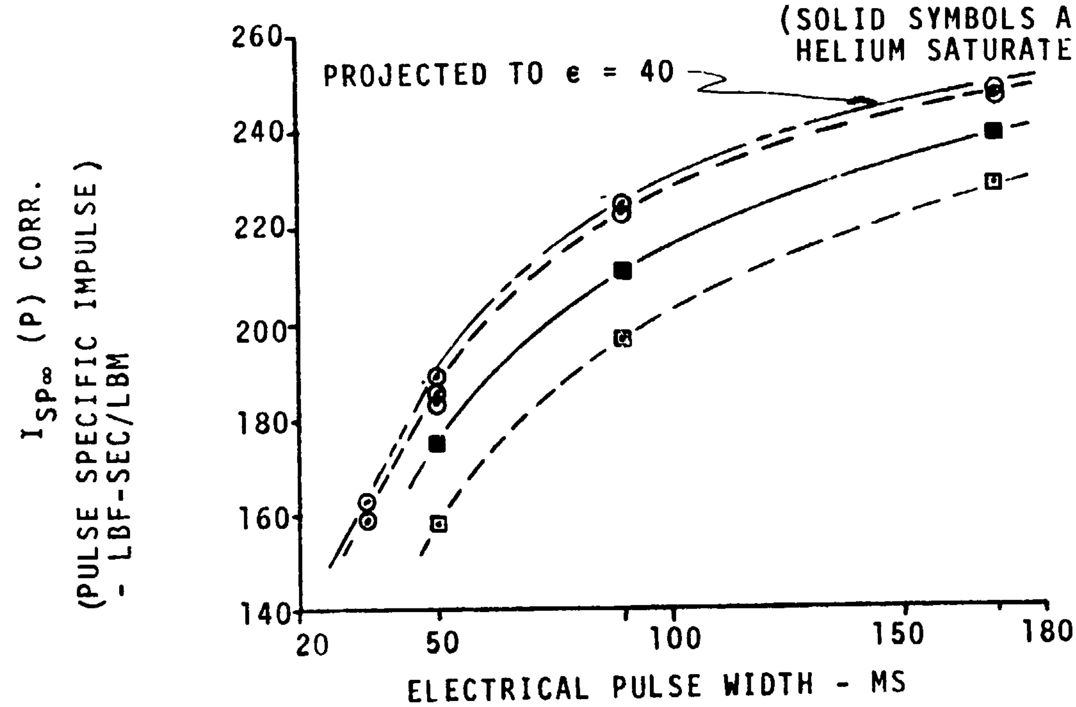
□ WITH TEST VALVE

△ WITH MOOG VALVE

○ INJECTOR S/N RDV-DD-PR-2 WITH MOOG VALVE

ϵ = 31

(SOLID SYMBOLS AND LINES INDICATE HELIUM SATURATED PROPELLANTS)



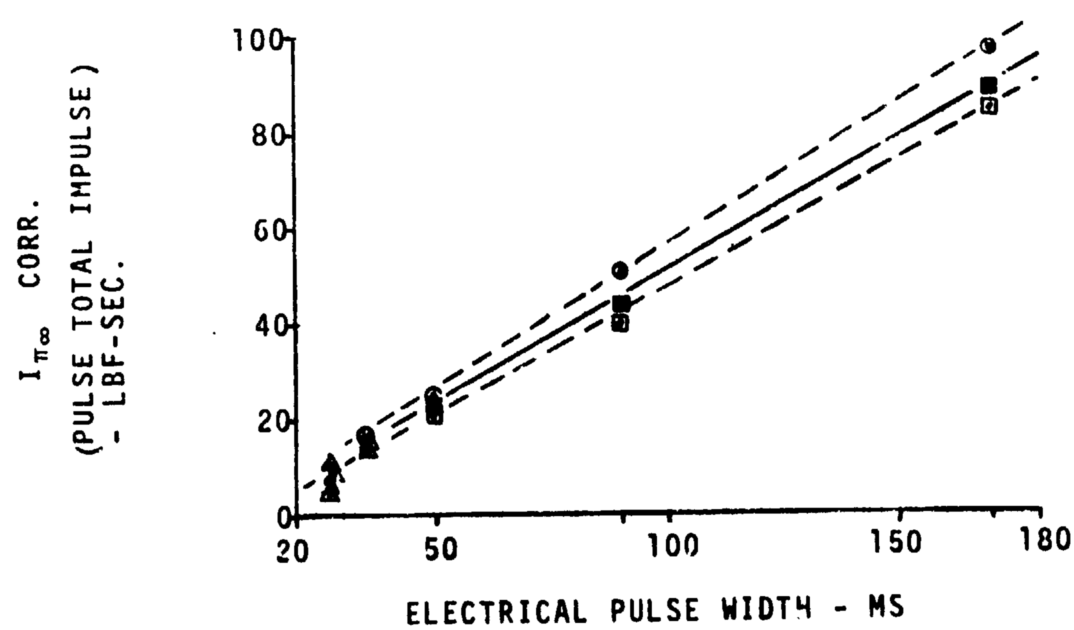


FIGURE 4.1-33

PULSING PERFORMANCE
VERSUS
PULSING FREQUENCY
(EPW = 50 MS)

STEADY STATE OPERATING CONDITION:
P_c = 2000PSIA, O/F = 1.6
PROPELLANT TEMPS. = 75°F

INJECTORS
□ S/N DD-PR-2 W/TEST VALVE
△ S/N DD-PR-2 W/MOOG VALVE
○ S/N RDV-DD-PR-2 W/MOOG VALVE

(SOLID LINES AND SYMBOLS INDICATE HELIUM SATURATED PROPELLANTS)

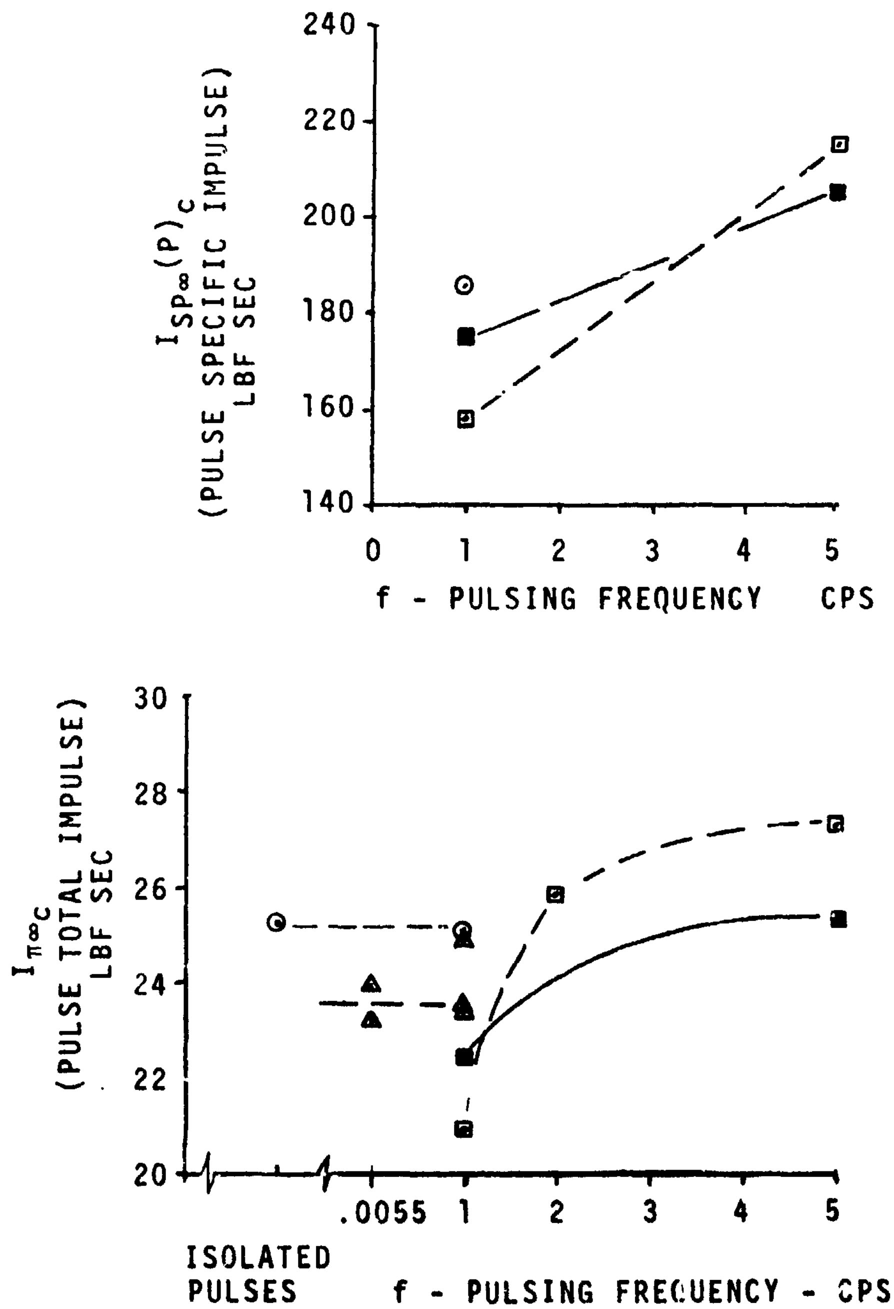
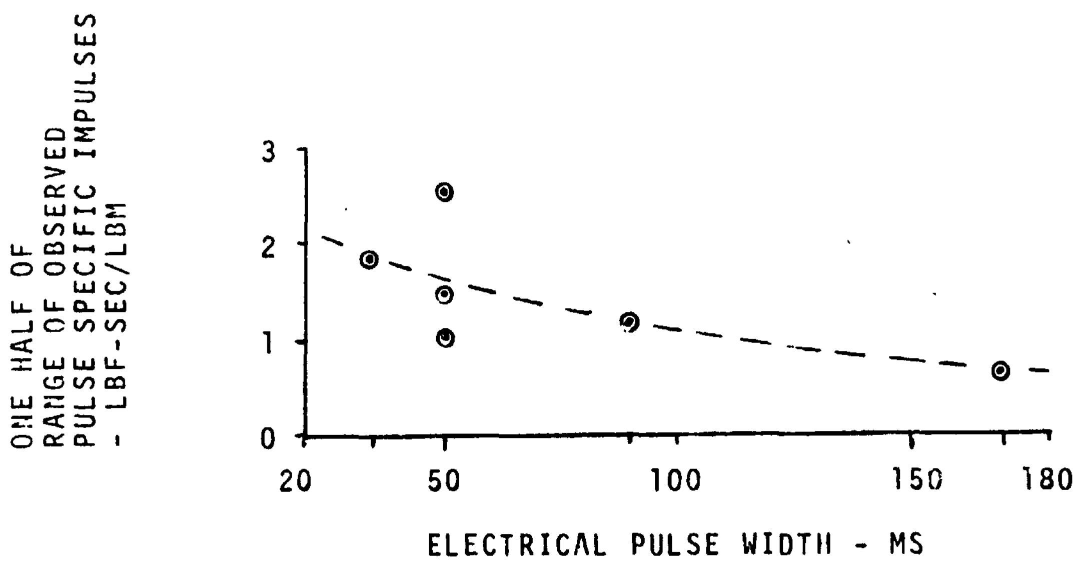


FIGURE 4.1-34

REPEATABILITY OF PULSING PERFORMANCE

f = 1 CPS

INJECTOR S/N RDV-DD-PR-2 WITH MOOG VALVE



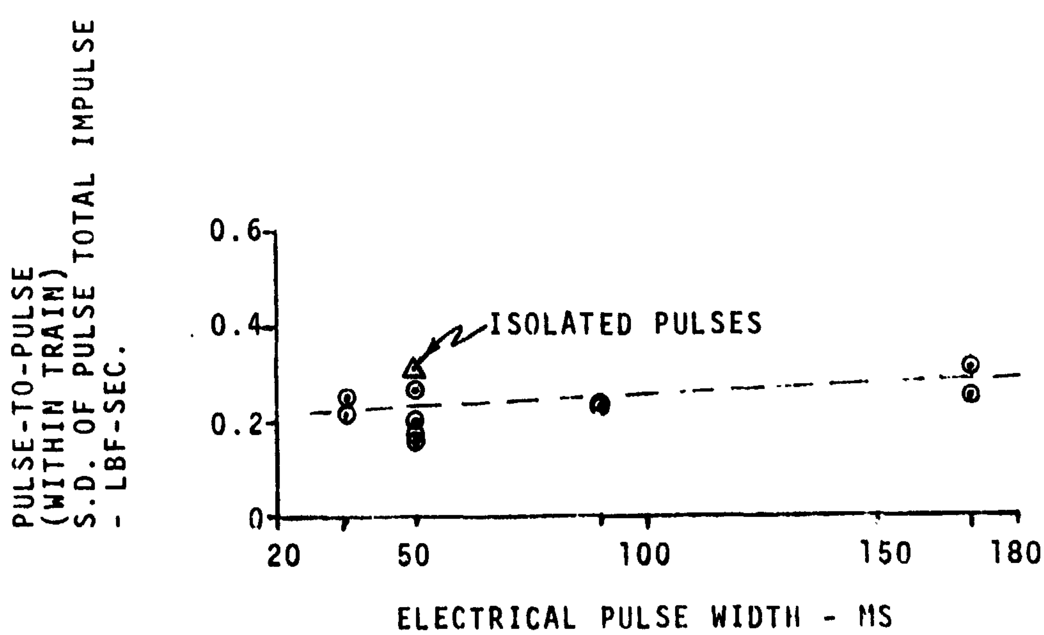


FIGURE 4.1-35

COMPARISON OF ANALYTICAL PREDICTIONS WITH TEST OBSERVATIONS

(RUN 6383-D3) - INJECTOR DDP2; TEST VALVE

TEST CONDITIONS: P_C = 200 psia

$R_{O/F}$ 1.6

P_A = 14.7 psia

TEMP = 70^oF

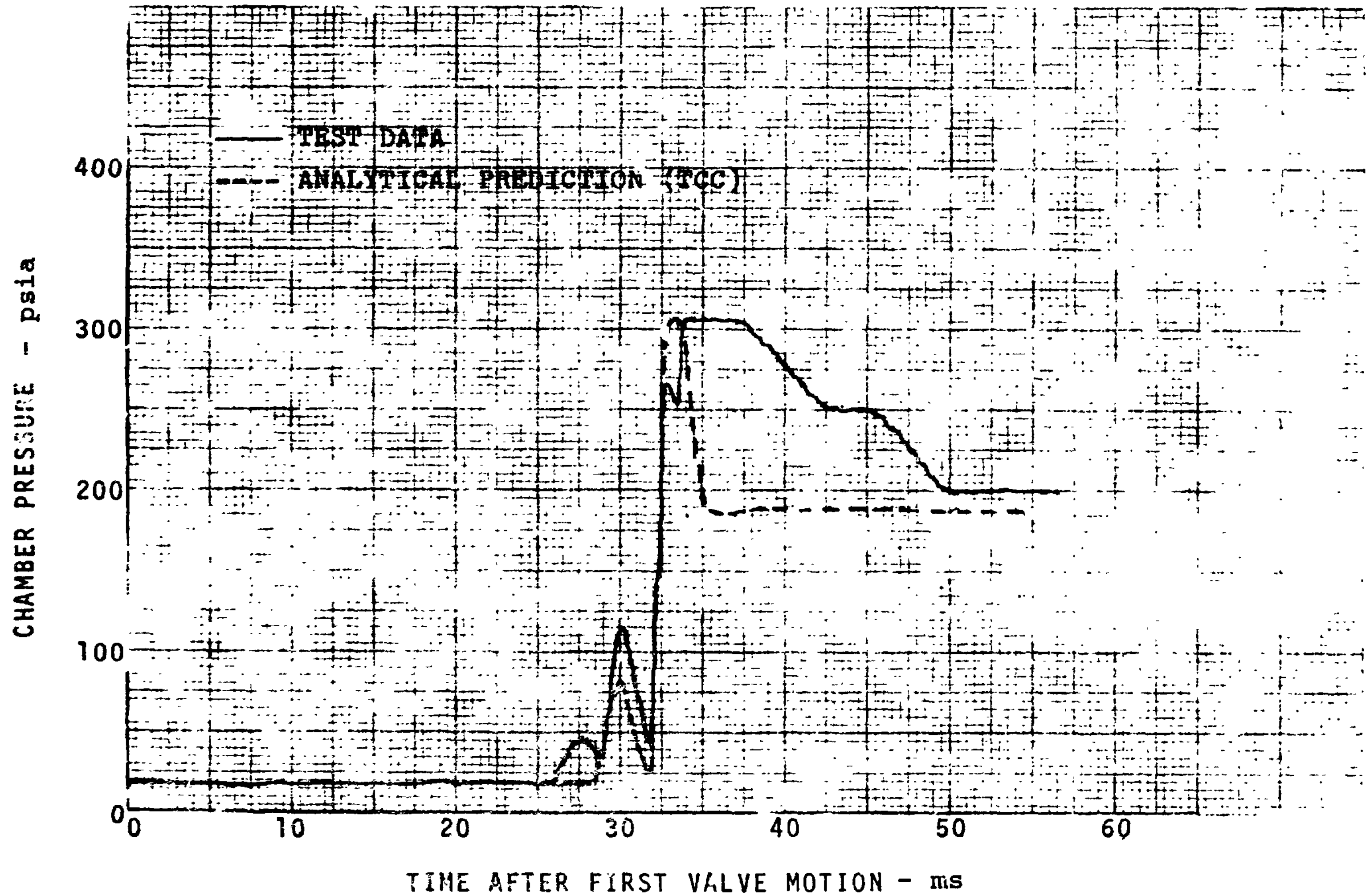
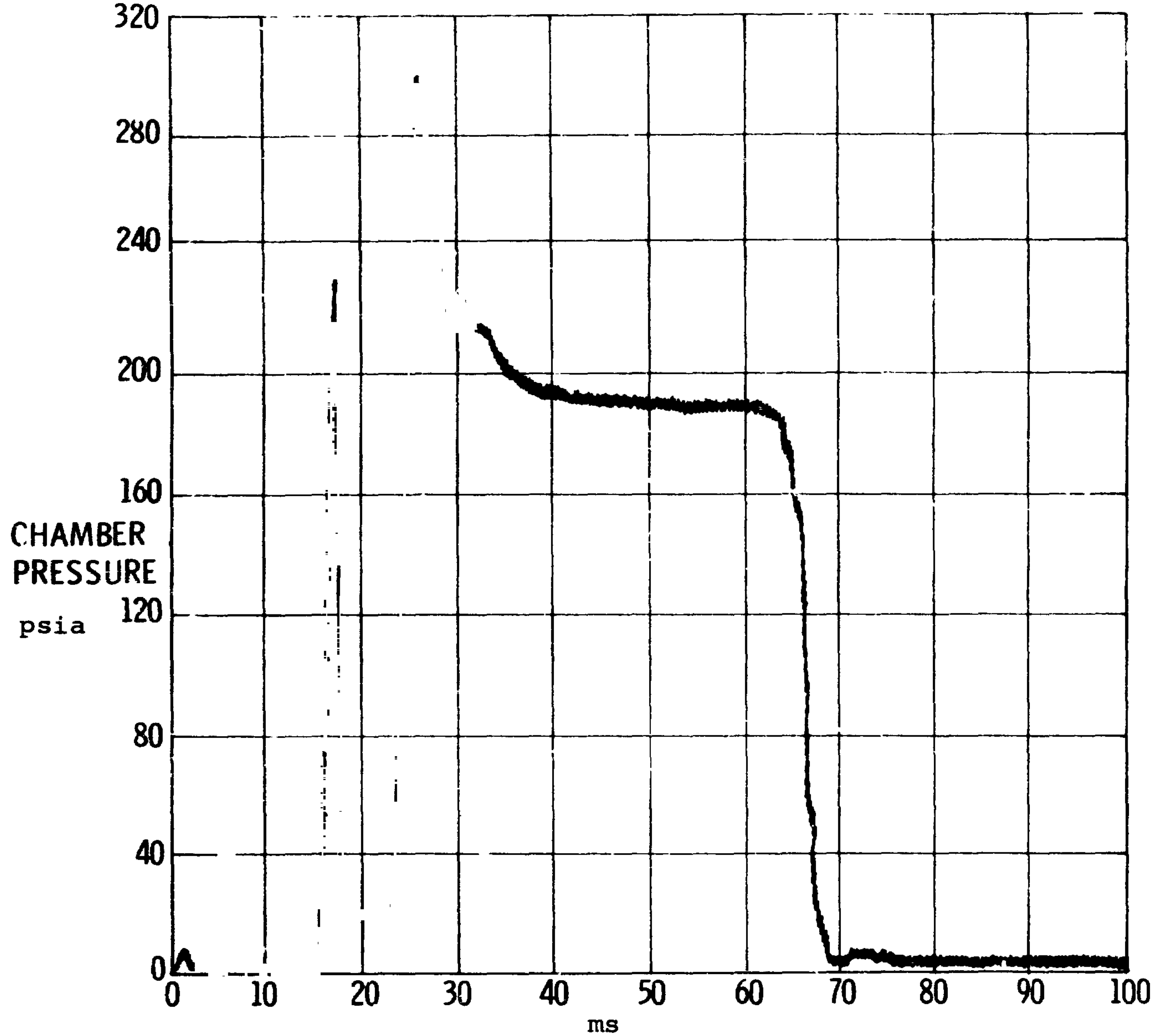


Figure 4.1-36

SIMULATED MINIMUM IMPULSE BIT - BASELINE ENGINE
(RDV INJECTOR - MOOG VALVE)



CONDITIONS:

$P_c = 200$

$R_{o/f} = 1.6$

PROPELLANT AND HARDWARE TEMP 70°F

DRY INJECTOR AND MANIFOLDS

$P_A = 0$

$\int F_\infty \, dt = 30$ lb-sec

Figure 4.1-37

COMPARISON OF CALCULATED AND TEST PULSING PERFORMANCE

$\epsilon = 31$

CURVES ARE TEST RESULTS AT f = 1 CPS AND WITHOUT HELIUM SATURATION

POINTS ARE MDAC TCC COMPUTER SIMULATION RESULTS

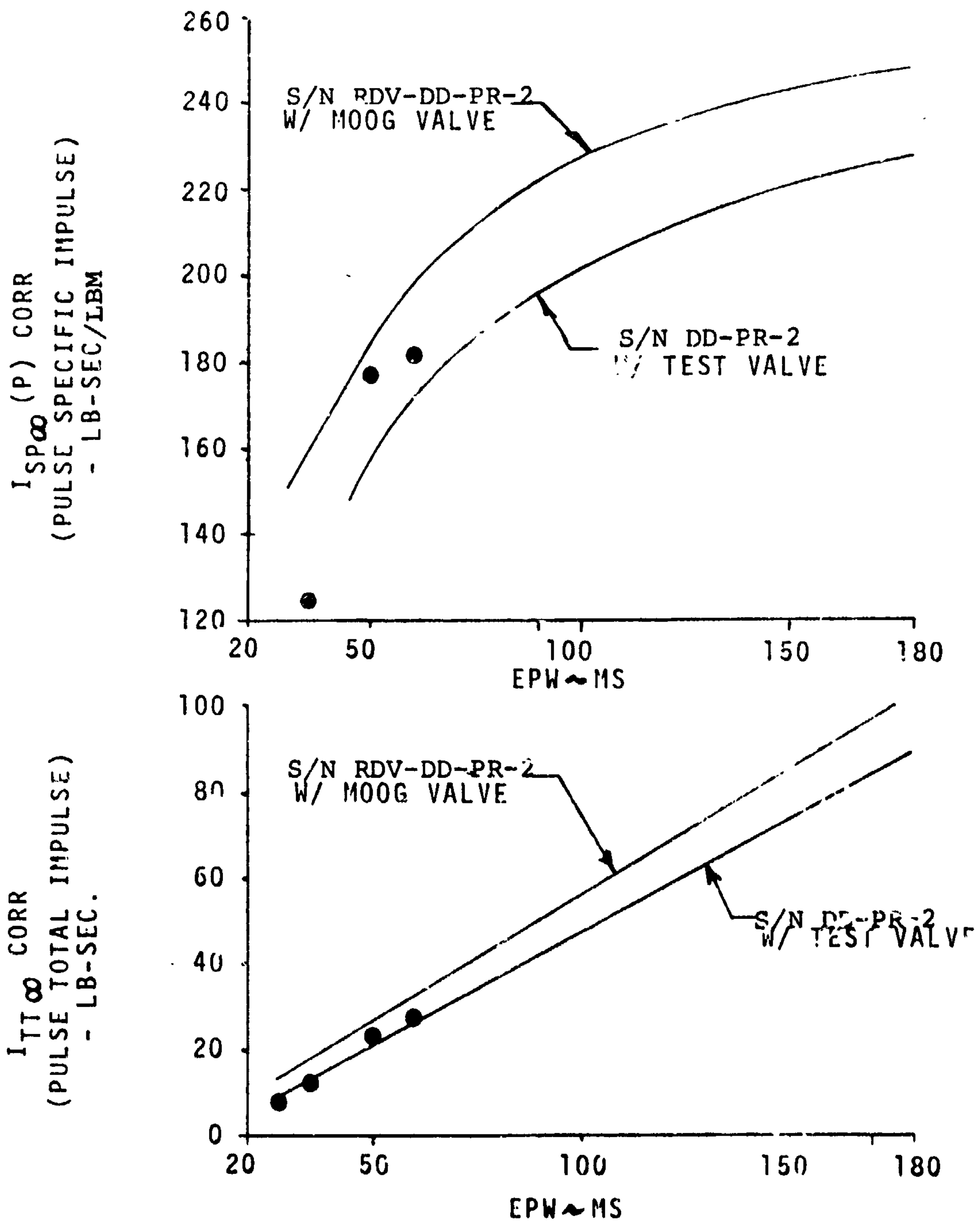


FIGURE 4.1-38

4.2 Phase II

This phase of the program consisted of the fabrication of two engine assemblies for the Phase III test program.

A design review was conducted in October 1972 prior to hardware release.

4.2.1 Phase III Test Hardware

4.2.1.1 Flight Type Engines

The flight-type engine (Figure 4.2-1) consists of a thrust chamber assembly (TCA), a bipropellant valve, and an external insulation assembly.

The TCA is a weldment of an injector assembly and a thrust chamber. The bipropellant valve is bolted to the thrust chamber assembly for easy replacement should the necessity arise. The preformed external insulation assembly is wire-laced to the thrust chamber assembly.

The flight-type engines are referred to by the designation of the injector assembly which forms their major element. Thus, the two engines tested during Phase III are referred to as engine S/N FT-2A and engine S/N RDV-2B.

4.2.1.1.1 Injector Assemblies

Two injectors were fabricated for the Phase III test program. Figure 4.2-2 shows the configuration of these units. The injector design was identical to the S/N RDY-DD-PR-2 test unit in regard to manifolding and impingement pattern although the mounting flange location was moved closer to the valve mounting plate. The injectors are of all-welded construction with 36 combustion doublets, 24 outer core doublets and 12 inner core doublets. The injectors contain a fuel vortex barrier for thrust chamber combustion zone cooling. Propellant manifold volumes have been minimized to enhance the pulse mode characteristics.

The orifice plate is fabricated of a columbium alloy (Cb-1Zr). The oxidizer manifold cover, fuel and oxidizer inlet tubes, the valve mounting plate support, and the valve mounting plate are titanium alloy. The flight-type injectors include titanium engine mounting points.

The injector incorporates acoustic cavities to allow the injector to operate in a dynamically stable condition throughout the engine operating regime.

The valve interface can be adapted to take either a test valve or a flight-type valve. The valve is bolted to the injector.

During the injector-level tests, the injectors were provided with flange and seal surfaces to mate with the test combustion chambers. These features were removed prior to TCA welding.

The individual injectors are described as follows:

S/N FT-1

This unit was fabricated during Phase II as a candidate for incorporation into a flight-type engine. The injection orifices were formed by conventional drilling. A fabrication error resulted in the drilling of the outer row of oxidizer orifices oversize which increased the outer core mixture ratio. Consequently, this injector was not utilized for Phase III engine testing.

S/N FT-2

Like S/N FT-1, this unit was fabricated specifically for Phase III. S/N FT-2 differs from the other injectors fabricated to date in that its injection orifices were formed by electro-discharge machining (EDM). Final barrier flow (ρ) for this configuration was 13.9%.

S/N RDV-2

Injector S/N RDV-DD-Pr-2 was built and tested during Phase I. It differs from the FT injectors only in its engine mounting provisions and chamber pressure pick-up configuration. (The engine mounting flange is fabricated of columbium alloy and is about 5/8 inch farther downstream than that of the FT injectors.)

Like FT-1, the RDV injector utilizes drilled orifices. Final barrier percentage was 13.5%.

4.2.1.1.2 Thrust Chambers

The flight-type thrust chamber (Figure 4.2-3) consists of a combustion chamber (fabricated of SCb-291) and a C103 nozzle extension which extends from $\epsilon = 5$ to $\epsilon = 40$. These pieces were welded together and coated with HiTemCo R512E silicide coating.

Prior to coating, 16 beads of weld were added to the outer chamber surface as shown in Figure 4.2-3. After coating, these beads were drilled to accept thermocouples.

4.2.1.1.3 Flight Type Bipropellant Valve

The flight-type bipropellant valve, a Moog Company design, is a normally closed, torque-motor-operated, bipropellant valve (see Figure 4.1-9).

The valve consists of a fuel propellant chamber, an oxidizer propellant chamber, and a torque motor. The fuel and oxidizer flapper buttons are attached to the armature of the torque motor by means of flapper rods. The flapper rods are supported by flexure tubes which act as flexible fluid barriers between the electromagnetic (torque motor) and propellant chambers of the valve.

4.2.1.1.4 External Insulation Assembly

During the testing with the columbium chamber and nozzle extension, the outer coated columbium surfaces of the engine were covered by a thermal insulation blanket of Dynaflex (12 lb/ft^3 density) with an outer covering of 0.005-inch titanium foil. The external insulation is mechanically attached to the engine.

4.2.1.1.5 Engine Modifications

During Phase III, engine S/N RDV-2B nozzle was reworked initially from $\epsilon = 40$ to $\epsilon = 35$ and subsequently to $\epsilon = 33$ to remove damage sustained in two facility incidents.

During simulated altitude testing, the engine fires into a water-cooled diffuser duct which connects to the steam ejector system to maintain altitude and remove exhaust gases from the engine nozzle expeditiously. On the first 100-second test of the worst case mission of Profile A burnthrough of the diffuser duct occurred due to inadequate cooling. As a result of this duct failure, the exit end of the nozzle was damaged resulting in cutting the nozzle back to $\epsilon = 35$. In preparation for the worst case mission of Profile C, the exit end of the engine nozzle was accidently damaged resulting in cutting the nozzle back to $\epsilon = 33$. All subsequent testing was conducted at this condition.

4.2.1.2 Non-Flight Type Hardware

During the injector level tests, non-flight-type hardware was used in several configurations. The most important of these is the bolt-together engine utilized for acceptance testing the injectors shown in Figure 4.1-2. The non-flight-type hardware is described as follows:

4.2.1.2.1 Combustion Chambers

S/N P-3

This thrust chamber consists of a columbium (SCb-291) liner with columbium (Cb-1Zr) flanges. It is coated inside and outside with HiTemCo R512E silicide coating. This chamber is provided with flanges that bolt to the injector at the upstream end and to a nozzle extension on the downstream end.

S/N B-1 (The Bomb Chamber)

The bomb chamber (Figure 4.1-11) is identical in interior contour to chamber S/N P-3, but has essentially no expansion beyond the throat. It is constructed of stainless steel and has ports for one flush-mounted and two water-cooled Kistler pressure transducers located 90° from each other. It also has a port for the insertion of the "bombs" and provision for the installation of accelerometers.

4.2.1.2.2 $\epsilon = 31$ Nozzle Extension

This unit was used during the bolt-together engine tests in the B-1 simulated altitude facility. It consists of a C103 nozzle section with a Cb-1Zr flange welded to it. The assembly is coated inside and out with R508C silicide coating and is designed to bolt to the S/N P-3 columbium thrust chamber. (See Figure 4.1-2)

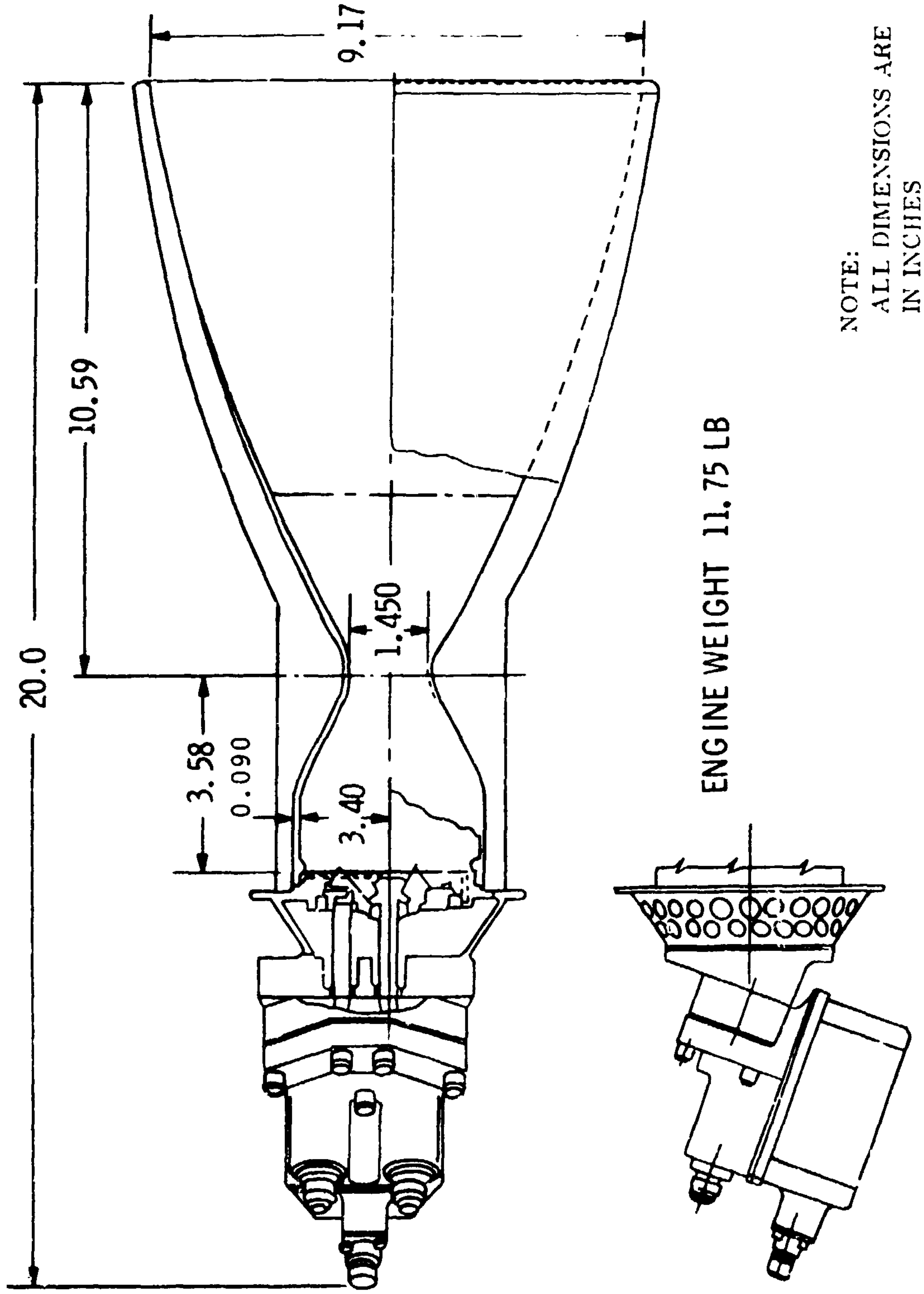
4.2.1.2.3 Test Valves

Two test bipropellant valves were available for use in situations where the flight-type valves were not available. These are normally open poppet type bipropellant valves.

The original valve was modified by adding a manifold so that the valve could be mounted directly to the injector and to keep the residual volume at a minimum.

4.2.1.2.4 Stability Test Bombs

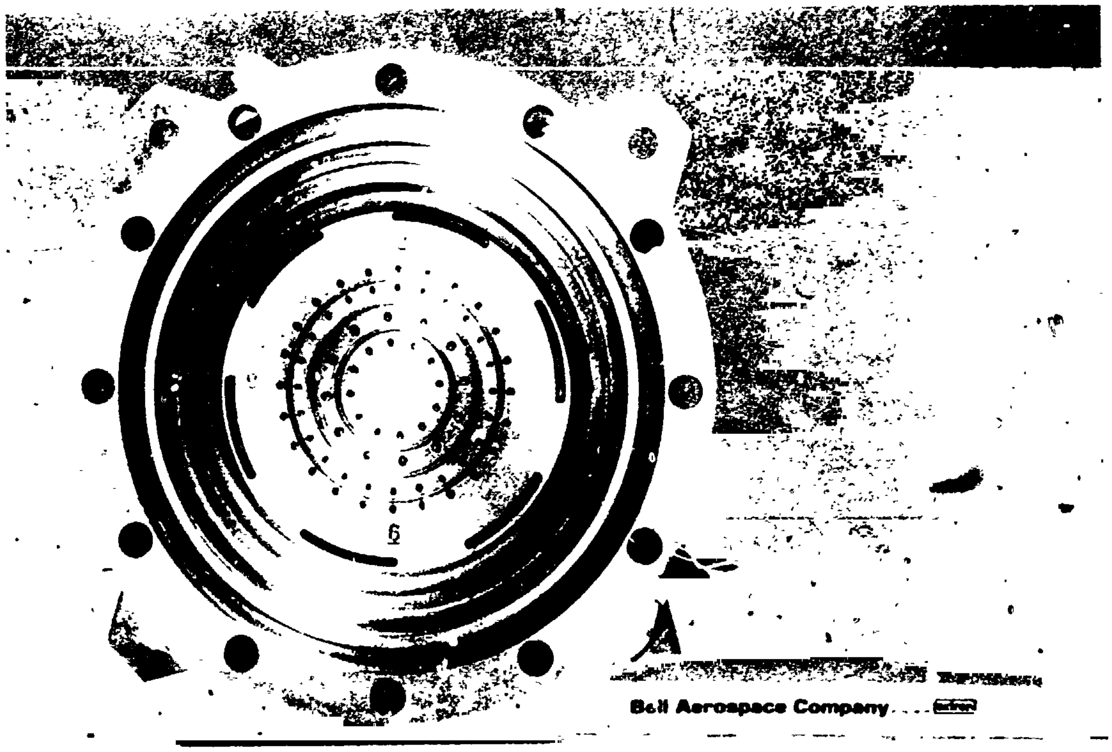
The explosive charge used to create chamber pressure spikes during the stability test series consists of a 2-grain bomb.



NOTE:
ALL DIMENSIONS ARE
IN INCHES

Figure 4.2-1 Flight - Type Engine

Bell Aerospace Company



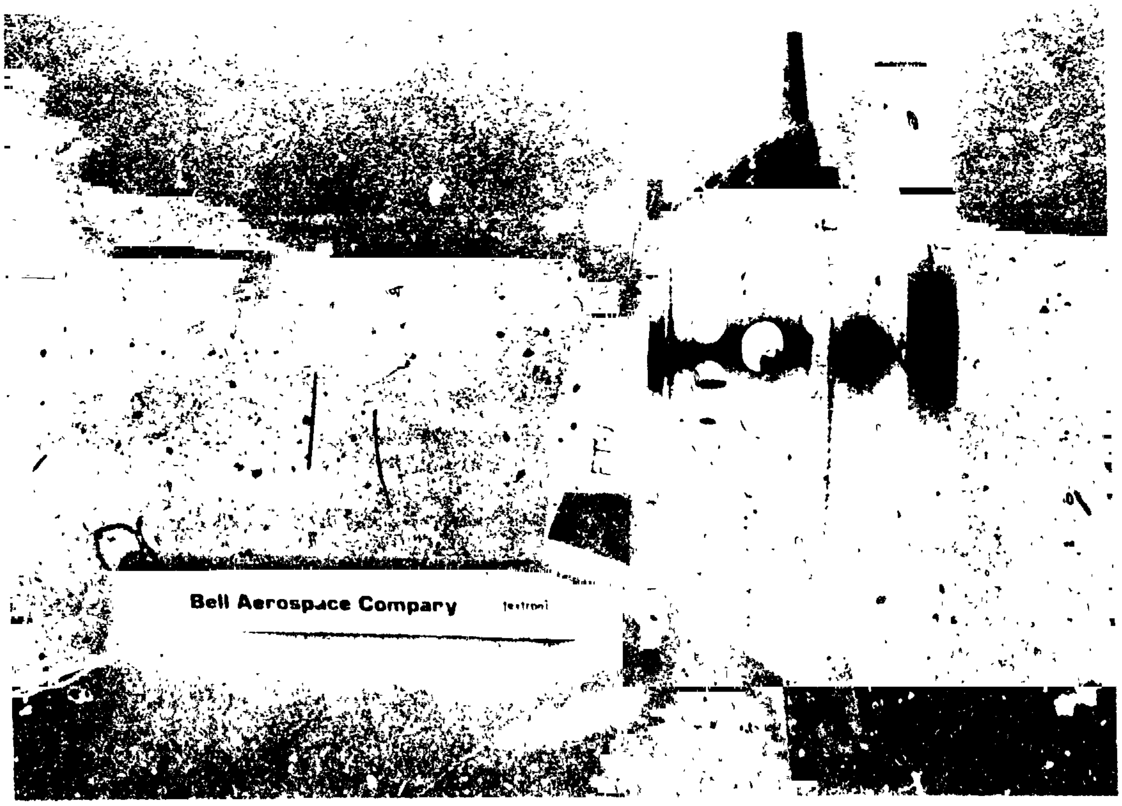


Figure 4.2-2 Flight-Type Injector

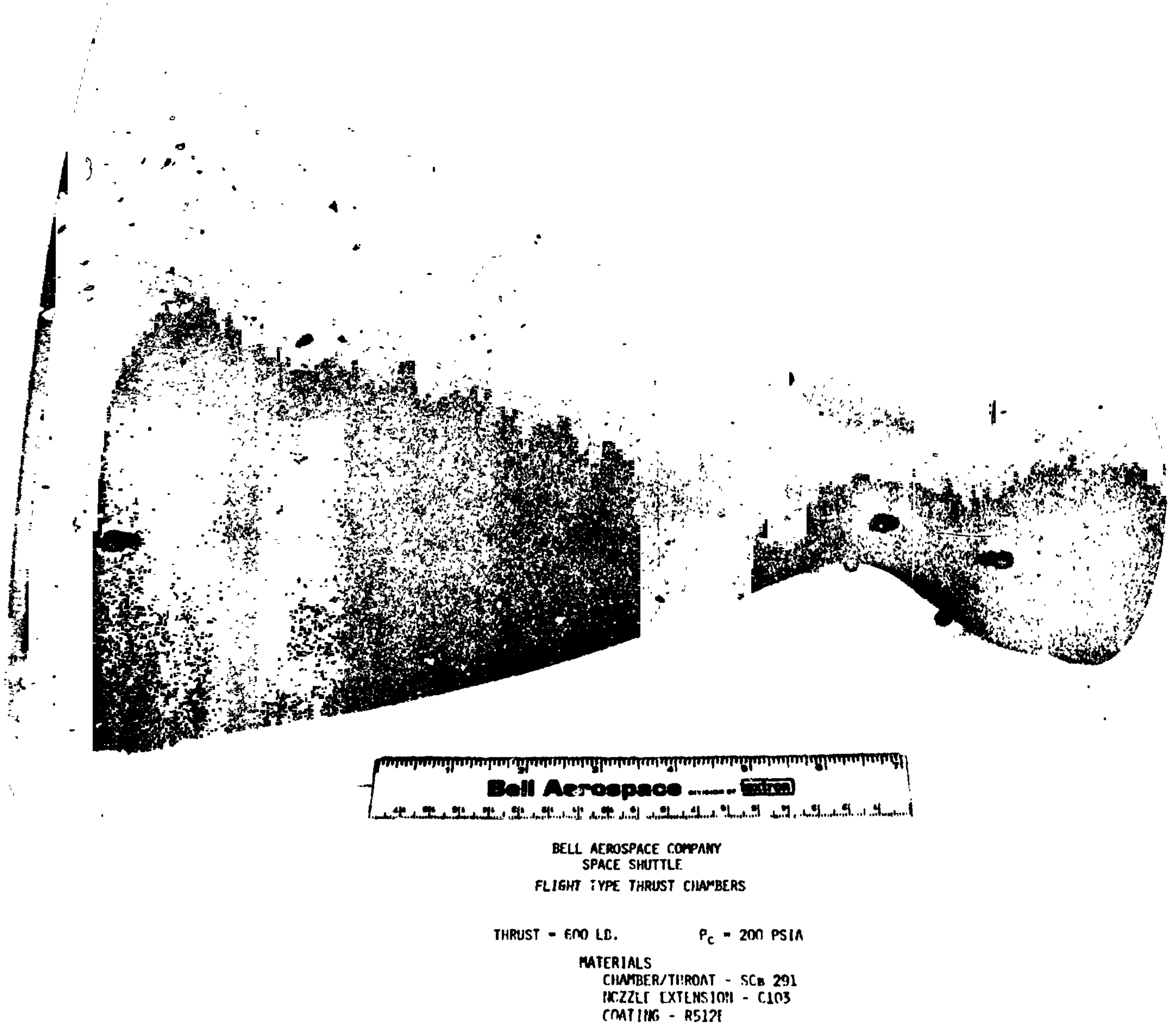


Figure 4.2-3 Flight-Type Thrust Chamber

4.3 Phase III

4.3.1 Test Program

Phase I was the first cycle of test, analysis and design leading to the fabrication of hardware during Phase II. Phase III consisted entirely of engine testing and the analysis of test data.

The Phase III test program was designed to demonstrate engine reusability with minimum maintenance and to simulate complete mission capability. Two flight-type engines were tested - each was to demonstrate a different aspect of the mission capability. Testing was initiated in February of 1973 and continued through early October 1973.

The test program (shown in Figure 4.3-1 and Table 4.3-1) included the following general areas:

- Injector cold flow
- Injector level tests (using bolt-together hardware)
    - Steady state performance
    - High altitude ignition
    - Bomb stability
- Engine level tests
    - Steady state acceptance tests
    - Environmental test engine (S/N FT-2A):
        - Composite environmental tests
            - Salt water spray
            - Sand and dust
            - Sinusoidal vibration
            - Humidity
            - Hot fire
        - Pulse mode fire tests
        - Worst case mission
        - Random vibration

TABLE 4.3-1

MISSION FIRING PROFILES

Profile A - Thermal Duty Cycle (Steady state firings + pulses)
Pulse Specific Impulse Tests
Worst Case Mission Duty Cycle (965 seconds on-time and 120 firings including 9-90 second firings + pulses).

Profile C - Same as Profile A except helium saturated propellants utilized.

Profile B - Endurance Test (600 seconds)

Profile D - Pulse Specific Impulse Tests and Steady State Performance Tests over Mixture Ratio Range of 1.5-1.7, Chamber Pressure Range of 180-220 psia and Propellant Temperature Range of 40-110°F.

Profile E - Modified Worst Case Mission Duty Cycle at maximum O/F, P_c and Propellant Temperatures with Helium Saturated Propellants.

Profile F - Sixteen Thermal Mission Duty Cycles (each consisting of 30 thermal cycle Steady State tests + 60 pulses accumulating approximately 360 seconds).

Note: Various duty cycles shown in Figure 4.3-2

- Hot fire test engine (S/N RDV-2B)
    - Multiple mission profile tests
        - Pulse mode
        - Steady state
        - Worst case and thermal
        - Endurance
    - Off design tests
        - Gas ingestion
        - Low chamber pressure
        - Off mixture ratio
    - Nozzle up tests
- Post test disassembly and inspection

The effects of helium saturated propellants were tested throughout the program. Post fire methods (maintenance and inspection) and valve cycle life data were also gathered.

A summary of the test data is presented in Appendix VI.

4.3.1.1 Test Facilities

1. Hot-Fire Test

Sea-Level Facility (D-3)

This test facility (also described in paragraph 4.1.1.1) is located at the Wheatfield Rocket Test Site and was utilized for the injector level testing (checkout and stability). The test stand, located in a reinforced concrete building, accommodates horizontal test firings of the engine. The test cell's propellant supply system is functionally similar to the steady state feed system of test cell B-1.

Altitude Test Cell (B-1)

Altitude testing of the engines was accomplished in Cell B-1 located in the altitude test complex of the Wheatfield Rocket Test Site.

Three test stand locations are provided within the 10 ft. diameter, 16 ft. high chamber. The three locations permit testing of engines under the following conditions:

a. Nozzle Down - MDC and Endurance Tests at 100,000 feet altitude. This position has thrust measurement capability (see Fig. 4.3-3).

b. Nozzle Down - Ignition tests at >250,000 ft. altitude. An extra stage was added to the steam ejection system which generates this high altitude.

c. Nozzle Up - Ignition tests at 100,000 ft. altitude.

The B-1 propellant system consists of two separate feed systems; MDC and propellant metering.

The MDC system was utilized for performance tests and MDC tests. The propellant metering system was utilized to determine pulse performance.

Thermal conditioning of the propellants is accomplished by means of a heat transfer fluid which is circulated through the propellant line jacket and also used to adjust the bulk temperature of the propellants.

Thermal conditioning of the injector-valve interface is accomplished by means of N_2 gas passing through a vortex tube.

2. <u>Environmental Test</u>

<u>Vibration Test Facility</u>

The Phase III vibration testing (Sine and Random) was performed in a vibration facility located in the BAC structures laboratory.

This facility consists of: an acoustic enclosure which contains the vibration units, a control console and a Data Recording and Analysis Laboratory.

The vibrators are one MB Electronics Model C-125 and one C-126. Either unit may be used with an oil film slip table.

The outputs of the test instruments are transmitted via coaxial cable to the Data Recording and Analysis Laboratory where they are appropriately conditioned before being recorded on a wide band magnetic tape recorder. The recorded outputs may be played back for analysis.

Salt Spray Exposure Facility

The salt spray exposure facility, is located in Building 7 at the BAC Wheatfield Plant. The unit is capable of providing salt spray under controlled conditions of brine flow, air flow, temperature and P_h in accordance with MIL-STD-810.

Sand and Dust Exposure Facility

Figure 4.3.4 is a schematic of the sand and dust exposure facility which is located in Building 7 at BAC's Wheatfield Plant. This unit is in compliance with MIL-STD-810.

In operation, a measured amount of the specified sand and dust is circulated through the test section by a controlled velocity stream of air.

Humidity Test Facility

Humidity testing was performed utilizing a Conrad, Inc. environmental chamber, Model FD-8-C-1-C. Humidity conditions are attained by means of programmed control of the wet and dry bulb temperatures.

4.3.1.2 Data Acquisition and Handling

Digital Data

The majority of test parameters necessary for calculating engine firing performance was recorded on one of two types of high speed Beckman analog to digital conversion systems: 6.9KHz for sea level testing in the "sea-level" facility and 30KHz for altitude testing, pulse mode evaluation and MDC's. These systems differ from one another in sampling rates and method of recording digitized data but have the same basic accuracy. This data was then processed by an IBM 360/44 general purpose digital computer, using programs developed by BAC.

Analog Data

Analog data was used to assess transient behavior of the engine such as combustion stability and ignition characteristics.

For transducer outputs having high frequency components, a wider band width is required than can presently be handled directly by the digital acquisition systems. In such instances, the raw data was converted to a wideband FM signal and recorded on magnetic tape via one of two Sangamo Model 4742 analog tape machines.

The playback output of the wide-band recorded system was directed via a programmable patchboard system to a wide-band (5 kHz) direct write oscillograph. This process allows time base expansion of the recorded traces.

Certain parameters were manually reduced from the resulting oscillograph records.

4.3.1.3 Testing

Phase III testing performed in accordance with the test matrix was basically a two engine test program. However, three injectors were tested - S/N FT-1, S/N FT-2 and S/N RDV-DD-PR-2.

Early in the Phase III test program a decision was made to increase thermal margin by increasing the percentage of fuel injected into the barrier. This resulted in several iterations - hot fire-rework-rehot fire cycles with each of the three injectors. After this, two injectors were selected (primarily based on schedule considerations) for assembly into engines. These two engines were identified by their injector serial numbers as S/N FT-2A and RDV-2B. Engine S/N FT-2A was primarily an environmental test unit while engine S/N RDV-2B was used to demonstrate fire test durability.

Engine S/N FT-2A (Figure 4.3-5)

Testing was initiated with two pulse trains performed, using the Propellant Metering System. These served to characterize the pulse mode performance of engine S/N FT-2A prior to the environmental tests. In addition, steady state firings were made to establish baseline performance.

The Composite Environmental Tests consisted of 10 exposures in the following sequence to:

- Salt spray
- Sand and dust
- Sinusoidal vibration
- Humidity
- Fire test

The Salt Spray test was conducted in the Bell salt spray exposure facility in accordance with MIL-STD-810B. The engine was mounted nozzle-up at a 45° angle simulating worst case vehicle geometry. Exposure time is 30 minutes per cycle. Figure 4.3-6 shows the engine in the test chamber.

Sand and Dust testing was performed (nozzle up attitude) in the Bell Sand and Dust facility (conforms to MIL-STD-810). Exposure was at 350 ft/min for 4 hours per cycle. The engine is shown in the facility in Figure 4.3-7 while Figure 4.3-8 shows the engine immediately after exposure

Sinusoidal vibration testing was performed in the Bell General Laboratory on two MB Electronics Model C-125 and C-126 vibrators as shown in Figure 4.3-9 and 4.3-10. The test sequence of 6 minute sweeps from 5 to 40 Hz at the specified input intensities of 1g from 5 to 23 Hz and an 0.036 inch D.A. from 23 to 40 Hz. Sweep rate was 1 octave per minute with additional 30-sec. dwells at the resonant frequencies.

For the 1st and tenth cycles the engine was vibrated in all three axes. Vibration was performed only in the worst case (X) axis (axial or longitudinal axis) for cycles 2 through 9.

Humidity testing was conducted in the Bell Rocket test area in Test Cell S-10. The exposure was to a temperature/humidity cycle consisting of a 2 hour dwell at 40°F and 95% humidity then 2 hours at a 160°F and 95% humidity with a 1 hour transient at the start and stop (from and to ambient) and between dwells.

The fire testing per cycle consisted of 25-50 ms pulses and then a steady state firing (5-sec).

Throughout the test series, no engine maintenance was performed. The fire test data demonstrated no change in engine operating characteristics (See Appendix VI, Part B).

Inspections performed using optical means and photography from the nozzle exit after the 1st, 5th, and 10th cycles (prior to hot fire), also showed no apparent hardware changes. Figure 4.3-11 shows the injector face prior to hot fire and Figure 4.3-12 shows the injector face after hot fire. Figure 4.3-13 also shows that there were no performance changes during the environmental/hot fire cycling.

Upon completion of the environmental cycling, the complete Post Flight Inspection was conducted and the Pulse Specific Impulse tests were repeated. No performance changes occurred.

The Worst Case Duty Cycle was then successfully conducted. Checkout tests were conducted to provide a baseline prior to random vibration.

C-2

The first test was to be a 25 minute (25 missions) exposure in the X (longitudinal) axis. The test was terminated early (23 minutes - 23 missions) due to indications that damage had occurred to the engine. Upon inspection, it was concluded that no further testing of Engine S/N FT-2A was possible. Inspection indicated a crack occurred in parent metal in the flange area external to the thrust chamber. Figure 4.3-14 shows the engine after random vibration. No further vibration was conducted. The engine was then subjected to disassembly and inspection.

Engine S/N RDV-2B (Figure 4.3-15)

The engine was acceptance tested and then subjected to the Missile Profile A test sequence. Worst case MDC test results from this profile are presented in Figure 4.3-16. During this test series the engine nozzle was damaged by a facility failure. (See 4.2.1.1.5) Rather than fire with an anomalous nozzle (and risk masking of potential problems) the damaged area was removed. This left the engine with a 35 to 1 expansion ratio nozzle. The engine was then subjected to Mission Profile C (Same as A but with He saturated Propellants). The continuous 600-second endurance run of Mission Profile B was then successfully conducted. Data from this test is presented in Figure 4.3-17. Prior to the 600-second test the engine nozzle was damaged again. This was due to an improper test installation. The damaged area was removed leaving the engine with a 33 to 1 expansion ratio. Mission Profiles D, E and F (with its 16 thermal duty cycles) were conducted with post flight inspections between each profile.

Mission Profile A which consists of a thermal duty cycle plus pulse specific impulse tests and a worst case mission duty cycle was intended to characterize the total performance of the engine at nominal operating conditions. Engine performance as a function of mission time for the Worst Case Mission Duty Cycle presented in Figure 4.3-16 indicates the engine operated with consistently high performance within a 1% band (around 295 sec. nominal). The maximum throat temperature was 2235°F. Data from the 600-second endurance test Run No. B-1-848 presented in Appendix VI was even more consistent in performance (0.3%) and thermal levels (2228°F max).

Mission Profile C which is the same as Profile A with helium saturated propellants yielded results very similar to the unsaturated tests in regard to steady state performance (and thermal characteristics) but showed improved performance on the 1 cps 50 ms pulses (I_{sp} = 210 sec. vs. 200 sec.).

Mission profile D was used to define pulse characteristics over a propellant temperature range from 40°F to 110°F with and without helium saturated propellants. As expected, the test results (See Figure 4.3-18) show a positive correlation with pulse performance ($I_{sp\infty}$ and I_T).

The worst case mission duty cycle (Mission Profile E) at maximum conditions (high O/F, P_c and T_p = 110°F) yielded a throat temperature about 50°F hotter (2300°F max) and performance about 1% higher (298 lb-sec/lbm) than nominal. This was considered to be better than the anticipated max. temperature of 2400°F to 2450°F.

Mission Profile F with its 16 thermal cycles including 4,600 firings and 1,344 seconds of firing time was intended to accumulate time and thermal cycles on the engine. The representative test mission duty cycle throat temperature profiles are presented in Figure 4.3-19. This specific part of the test program lasted approximately two weeks (test time). The performance level during that two week period stayed at 295 lb-sec/lbm, within a ±0.5%, band as indicated by the data presented in Appendix VI.

With the successful completion of the specified mission firing profiles, the engine was then subjected to the off design tests.

Off Design Tests (Appendix VI, Part D)

Four types of off design tests were conducted as follows:

a. Helium bubble ingestion tests

   Oxidizer side only
   Fuel side only
   Oxidizer and fuel sides simultaneously

b. Low chamber pressure tests

   Nominal chamber pressure of 200 psia
   Low chamber pressure
     180 psia
     150 psia
     120 psia

c. Off mixture ratio tests

   Low mixture ratio (O/F = 1.0) at nominal chamber pressure
   High mixture ratio (O/F = 2.2) at nominal chamber pressure

d. Nozzle up tests

90 ms pulses with saturated low temperature propellants
50 ms pulses with saturated low temperature propellants.

Helium Bubble Ingestion Tests

Six tests were successfully conducted at altitude wherein helium gas bubbles were introduced just upstream of the engine with the helium flow rate 10% by volume of the total propellant flow rate. Special instrumentation was utilized for these tests (as well as the other types of off design tests) to determine if low frequency oscillations were present. High frequency Kistler transducers were utilized to measure the oxidizer and fuel feed pressures to the engine, and one chamber pressure transducer attached directly to the engine was utilized. The results were recorded on FM tape. The results (Part D of Appendix VI) indicate no change in engine performance or wall temperature during these helium bubble ingestion tests and no low frequency oscillations in chamber pressure or feed pressures. One test (B-1-955) indicated an oxidizer feed pressure oscillation of ±5% during helium flow which is within the requirements. Three of the tests were 60-seconds in duration to evaluate engine durability and indicated no anomalies.

Low Chamber Pressure Tests

Four tests were conducted to determine the capability of the engine to withstand a system malfunction wherein the engine would operate at low chamber pressure. The same special instrumentation discussed above was utilized. The results (Part D of Appendix VI) indicate satisfactory engine operation to 74% of nominal chamber pressure. However, at very low chamber pressure (57% of nominal) there appeared to be a 9% reduction in engine performance with low wall temperatures ($<1600^{\circ}F$). There are low frequency oscillations (±25% in chamber pressure) associated with very low injector pressure drop at this condition. Engine durability was not compromised in any manner even at very low chamber pressure.

Off Mixture Ratio Tests

A test series was conducted to determine the capability of the engine to withstand a system malfunction wherein the mixture varied widely from nominal (O/F = 1.6). The nominal total propellant flow rate was utilized for these tests as a worst case condition although a system malfunction would normally be at a lower total flow rate due to system blockage (regulator, valves, etc.). The data is presented in Part D in Appendix VI. The low mixture ratio (O/F = 0.97) tests indicate a decrease in performance (7%) with very low wall temperatures (<1000°F). The high mixture ratio (O/F = 2.2) tests indicate wall temperatures (2300°F) similar to the worst case mission duty cycle at maximum operating conditions (maximum P_c, maximum O/F, and helium saturated propellants at 110°F) with a performance decrease of 5.5%. Consequently, engine durability was again successfully demonstrated.

Nozzle Up Tests

A series of nozzle up tests was conducted at simulated altitude to determine the potential for manifold explosions. No manifold explosions occurred and the tests were completely successful and without incident; however, there was an indication of seat leakage as measured with GN_2 although no liquid leakage. The following tests were conducted:

15-90 ms pulses at 5 cps
15-90 ms pulses at 1 cps
15-90 ms pulses at 0.2 cps
15-50 ms pulses at 5 cps
15-50 ms pulses at 1 cps
15-50 ms pulses at 0.2 cps

The same special instrumentation presented above was utilized. All tests were conducted with 40°F helium saturated propellants. Consequently, the design demonstrated the capability of the engine to successfully fulfill all presently known firing requirements of the Space Shuttle.

Valve Testing

Phase III testing included evaluation of three flight type bipropellant valves (S/N 001, S/N 002 and S/N 003). These valves were identical to the Phase I prototype valves with some minor seat design changes intended to improve seating characteristics.

Another design change made was the incorporation of vibration isolators to minimize wear of the teflon seat during lateral vibration. The vibration isolators incorporated into the flight-type design enabled valve S/N 003 to withstand the random vibration tests to which it was subjected.

The three flight-type valves were received at Bell during the latter part of March 1973.

S/N 001

This valve was initially assembled to the RDV-2B engine. However, the valve would not open with 310 psig inlet feed pressure applied, making fire testing impossible. The valve was removed and replaced with S/N 002.

Valve S/N 001, was tested at the valve level for opening response times at various inlet pressures. Results indicated the valve was sensitive to inlet feed pressures of 300 psig and greater.

The valve was returned to Moog for evaluation. The bias of the torque motor was changed slightly to increase the opening force margin.

The valve was returned to Bell and released to stock to be used as a backup valve for S/N 002 or S/N 003.

S/N 002

As indicated above this valve was installed on Engine S/N RDV-2B for the hot fire test program. During this porgram the valve was subjected to 6377 engine starts and to 10,411 seconds of firing time.

Throughout the engine test program. no engine (including valve) maintenance was conducted. During the multi-mission hot fire test program, the valve operated satisfactorily for all starts with the exception of high chamber pressure tests. Because of the high feed pressures required, the coil power was not sufficient to overcome the high unbalance force. Seat leakage rates were satisfactory and no indication of external leakage was ever observed.

Upon completion of the nozzle up tests (end of fire test program) the valve was tested for opening response times and for leakage. Response times were acceptable but seat leakage was anomalous as presented in Table 4.3-2.

The valve was then returned to Moog for evaluation. A complete A/T was conducted with only seat leakages being out of specification.

TABLE 4.3-2 - VALVE TESTING AT BAC

Valve S/N	Date(1973)	Seat Leakage ($CC\ GN_2/Hr$) @3 psig Fuel	@3 psig Ox	@300 psig Fuel	@300 psig Ox	Comment
001	March	0	0	0	0	Before Cycling
		0	0	2.5	0	After Cycling
	June	0	0	0	0	Moog After Rework
002 (Engine S/N RDV-2B)	March	0	0	0	0	Before Cycling
		0	0	0	0	After Cycling
	6/12	0	0	0	0	Pre Engine A/T
	8/14	0	0	0	0	Post-Mission Profile B
	9/7	420 (Both)		30 (Both)		Post TMDC No. 12.
		10 (Both)**		0 (Both)		
	9/17	0	0	0	0	Post Mission Profile F
	9/28	90 (Both)		0	0	Post Off Design Tests
	10/5	0	90	0	720	Post Nozzle Up Tests
003 (Engine S/N FT-2A)	March	0	0	0	0	Before Cycling
		0	0	0	0	After Cycling
	6/11	0	0	0	0	Pre Engine A/T
	6/21	0	0	0	0	Pre Comp. Env. Tests
	7/30	0	0	0	0	Post Exp. #5
	8/15	0	0	0	0	Post Comp. Env. Tests
	8/27	0	0	0	0	Pre Random Vibration
	Sept.	0	0	0	0	Post Random Vibration

*300 psig at inlets.
**Valve removed, re-installed and rechecked.

The valve outlet manifolds were removed and the seals and nozzles were inspected. Many very fine contaminant particles (too small to analyze) were found in the flow sections of the valves (including filter inlets) and on the seats. Salts were also found on the oxidizer side. These very fine particles appeared to come from the upstream test system.

The leakage across the oxidizer side was probably due to a contaminant lodging on the valve seat during the nozzle up tests. The contaminant was subsequently washed away during flushing and purging prior to disassembly and inspection. However, the leakage rate was not excessive and no liquid leakage was noted.

S/N 003

This valve was assembled on Engine S/N FT-2A for its multi-mission environmental/hot fire test program. During this program, the valve was subjected to 1976 engine starts and 1230 seconds of firing time in addition to its multi-mission environmental cycling of salt water spray, sand and dust, vibration and humidity. As with valve S/N 002, no maintenance was performed on the unit during these tests. The valve operated satisfactorily throughout the test program with no anomalies noted. Table 4.3-2 presents the leakage data.

4.3.1.4 Phase III Summary of Results

The design of the Space Shuttle RCS Engine has the primary objective of reusability with minimum maintenance. The two engines of Phase III have demonstrated this objective by successfully completing a multi-mission environmental/hot fire test program with Engine S/N FT-2A and a multi-mission hot fire test program with Engine S/N RDV-2B with no maintenance.

Engine S/N RDV-2B successfully completed its entire hot fire test program including off design tests of helium bubble ingestion tests, low chamber pressure tests, off mixture ratio tests and nozzle up tests. The engine has consequently demonstrated the capability of fulfilling all known firing requirements of the Space Shuttle. The engine accumulated the following life:

	Total Life (No Maintenance)
Firing Time	10,411*Seconds
Firings	6,377
Thermal Cycles	567

Engine S/N FT-2A successfully completed its environmental/hot-fire test program which included ten cycles of:

1. Salt Spray
2. Sand and Dust
3. Sinusoidal Vibration
4. Temperature/Humidity
5. Hot-Fire

The engine test program was conducted with no engine maintenance.

During random vibration after the environmental/hot fire cycling, the engine failed in the injector flange area after the equivalent of 23 mission cycles due to high amplification factors. Analysis of the problem has indicated utilization of dampers at the mounting points will correct the problem. It may also be noted that the RI/SD requirements (0.7 g^2/Hz maximum) for random vibration are significantly less than the technology requirements (1.6 g^2/Hz maximum).

Steady State Performance

The steady state performance goal of 295 lb-sec/lbm was demonstrated on all three prototype test units. Engine thermal characteristics were as predicted - approximately 300°F - 400°F in the thrust chamber with a maximum of 2250°F in the throat. The performance characteristics were stable over the mixture ratio and chamber pressure range tested. Tests with helium saturated propellants indicated that performance was not impacted by saturation. A summary of the steady state performance correlations is shown in Figure 4.3-20.

Combustion Stability

Twelve successful bomb tests with both Flight Type (FT) injectors (6 each) demonstrated attainment of the dynamic stability criteria. The results are shown in Appendix VI including a typical oscillograph trace from FM tape.

*Over ten missions as defined by the technology contract but over 50 missions as defined by RI/SD (200 seconds/mission).

High Altitude Ignition Tests

High altitude (>250,000 ft) ignition tests at low (40°F) and high (+120°F) propellants were successfully conducted with both Phase III engines. The maximum spike level recorded was 740 psia. The results are shown in Appendix VI including an oscillograph trace from FM tape of one of the highest spikes (720 psia) obtained.

Pulse Mode Performance

Pulse mode operational characteristics were defined for both Phase III engines. The test results (correlated with appropriate pulse mode parameters) are presented in Figures 4.3-21, -22, and -23. An FM tape playback is shown for various pulses through a train in Figure 4.3-24 showing the excellent reproducibility obtained.

The preliminary results of Phase I were confirmed by the Phase III testing. Summarized, they are as follows:

a. The goal of 220 lb-sec/lbm specific impulse at a EPW of 50 ms and 5 cps was demonstrated.

b. Increased propellant temperature improves both pulse specific impulse and impulse bits.

c. Helium saturation of propellants generally improves pulse performance.

d. Increased pulse frequency improves pulse specific impulse but has little affect on impulse bit. The effect of propellant saturation on pulse performance is dependent on pulse frequency.

Flight Type Valve

The main problem encountered with the dual semi-balanced seat design valve during the Phase I portion of the program was seat leakage. To overcome this problem for the Phase III portion of the program, several minor seat design changes were made.

The Phase III results showed these changes were effective in reducing leakage to specification limits (maximum of 10 sec/hr N_2 Gas at 3 to 300 psig), but another problem became apparent. This was the sensitivity of the valve to inlet pressures above 300 psig. (Although the design requirements specify a static feed pressure of only 300 ± 6 psia, actual testing conditions for malfunction type test demonstrations require the valve to operate with feed pressures up to 340 psia.)

Minor modifications will allow the valve to still meet leakage requirements but not be sensitive to feed pressures up to 350 psia.

Maintenance and Inspection

The two Phase III engines each demonstrated the capability of 10 equivalent Space Shuttle missions* without any engine maintenance. Engine S/N FT-2 was subjected to 10 environmental/hot fire cycles with no maintenance. Engine S/N RDV-2B was subjected to 10,411 seconds of firing in 6377 starts with no maintenance. It is thus concluded the columbium engine requires no routine maintenance for the RCS application. In addition, the visual inspections (with appropriate optical aids) is sufficient to determine thrust chamber flight worthiness (i.e., coating, thrust chamber, injector orifice plugging, etc.)

Life

The Bell Columbium RCS Engine has demonstrated a 10 mission - 10,000* seconds life capability. The engine disassembly and inspection results (presented below) show no engine degradation with the 10,000 seconds of firing. Thus it is concluded the design is capable of meeting the Space Shuttle RCS life requirements.

4.3.2 Disassembly and Inspection

The two flight-type engines (S/N FT-2A and S/N RDV-2B) were subjected to disassembly and inspection. The objectives were to determine the impact of the multi-mission environmental/hot fire test program on S/N FT-2A and the multi-mission hot fire test program on S/N RDV-2B, both engine test programs conducted with no maintenance.

4.3.2.1 Engine S/N FT-2A

The injector of engine S/N FT-2A was disassembled from the engine and inspected by metallurgical sectioning and evaluations. Figure 4.3-25 shows a cross-section of the injector indicating sound electron beam welds and no contamination due to the multi-mission environmental/hot fire test program. Figure 4.3-26 shows a typical photomicrograph of the microstructure of the injector face. Only a very slight (0.004-0.006

*Equivalent to 50 missions as defined by RI/SD.

inch max.) oxide layer was exhibited and confirmed by hardness traverses. The conclusion reached was that no problems exist even with no maintenance. In addition no embrittlement was experienced or exhibited.

4.3.2.2 Engine S/N RDV-2B

Engine S/N RDV-2B was cut into a longitudinal section and inspected by metallurgical sectioning and evaluations. Figure 4.3-27 shows the section of the injector face. The engine completed its entire comprehensive multi-mission hot fire test program accumulating 10,411 seconds at the engine level. The injector saw 10,738 seconds firing time due to additional injector acceptance tests.

Photomicrographs were taken of the injector (S/N RDV-2) to examine the microstructure. Figure 4.3-28 shows the central region of the injector face,which is typical,indicating a very slight oxide layer which was also confirmed by hardness traverses. The depth of the oxide layer (0.005" max.) was the same as indicated on injector S/N FT-2A even though there was eight times the firing time on S/N RDV-2B.

Photomicrographs were taken of the thrust chamber (S/N RDV-2B) to determine the coating condition. Figure 4.3-29 shows the complete engine section and Figure 4.3-30 shows the coating microstructure throughout the chamber including the multiple phases of the coating and indicates 0.003 inch coating thickness remaining after 10,411 seconds firing time and 567 thermal cycles compared to approximately 0.004 inch initially. (For comparison an unexposed coated SCb-291 sample strip is presented in Figure 4.3-31).

Tensile bars were made from the circumferential section of the chamber and nozzle extension to determine the ductility and embrittlement characteristics. The results indicated excellent ductility, elongation and tensile results similar to the original material data.

No embrittlement was experienced or exhibited on the injector or thrust chamber.

The conclusion reached from the disassembly and inspection of engine S/N RDV-2B is that the engine will fulfill the Space Shuttle known firing requirements with no maintenance.

4.3.2.3 Disassembly and Inspection Conclusions

The Bell Columbium Vortex Cooled Engine will meet the Space Shuttle RCS engine requirements. No embrittlement was experienced or exhibited on either injector S/N FT-2A or injector S/N RDV-2B which had eight times more firing time than S/N FT-2A. The coating of the S/N RDV-2B thrust chamber was in excellent condition and the thrust chamber exhibited no embrittlement but excellent ductility and elongation even after 10,411 seconds firing time and 567 thermal cycles.

PHASE III TEST MATRIX

	TEST UNIT		
	FT-1	FT-2	RDV-2
Injector Cold Flow	X	X	X
Performance	X	X	X
High Altitude Ignition	-	X	X
Bomb Stability	X	X	-
Engine Acceptance	-	X	X
Environmental			
a. Salt Spray	-	X	-
b. Sand and Dust	-	X	-
c. Sinusoidal Vibration	-	X	-
d. Humidity	-	X	-
e. Random Vibration	-	X	-
Mission Fire Tests			
a. Pulse Mode	-	X	X
b. Steady State	-	X	X
c. Worst Case	-	X	X
d. Endurance	-	-	X
Off-Design			
a. Gas Ingestion	-	-	X
b. Low Chamber Pressure		-	X
c. Off Mixture Ratio		-	X
Nozzle Up	-	-	X
Post Test D&I	-	X	X

FIGURE 4.3-1

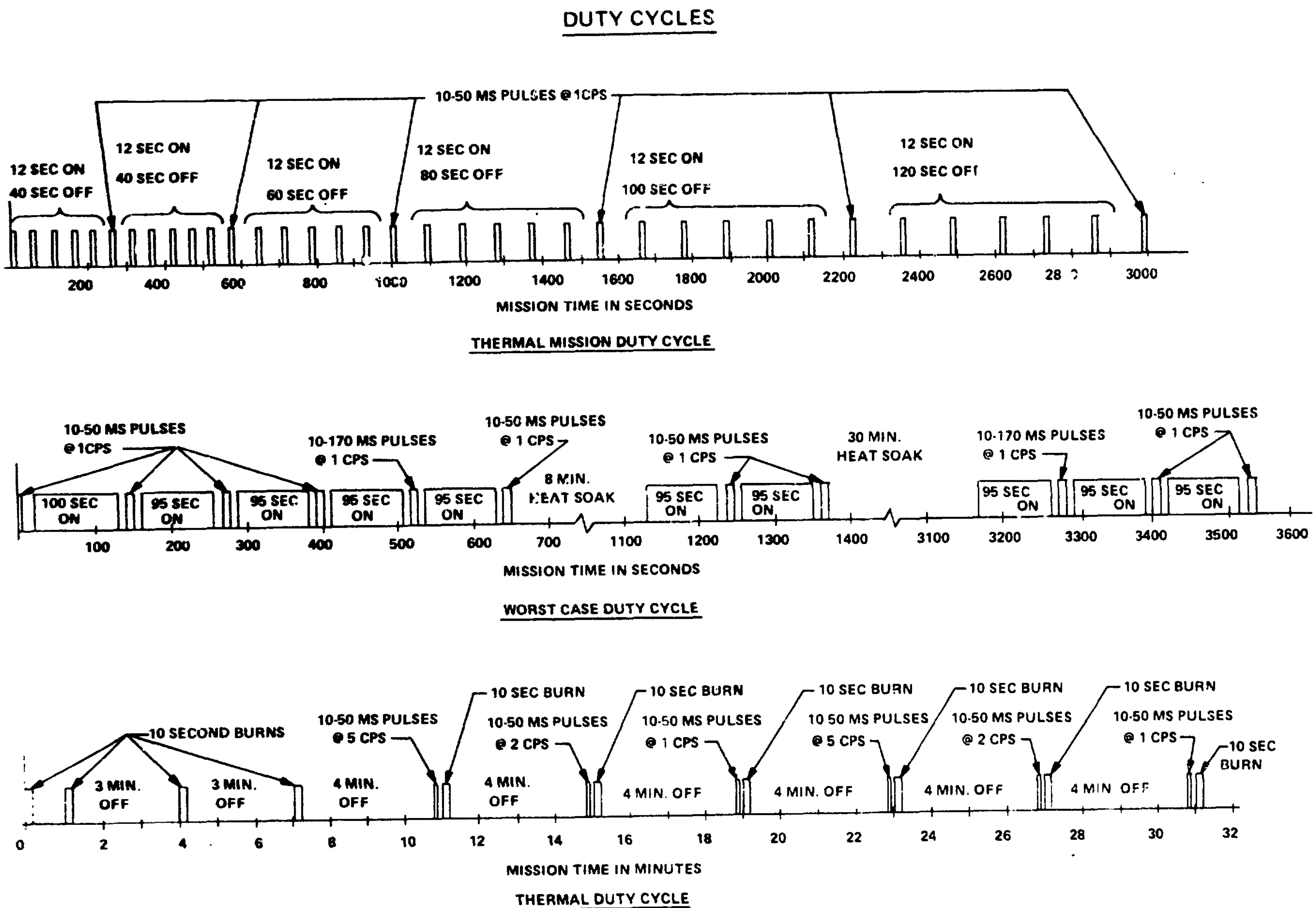

DUTY CYCLES
10-50 MS PULSES @1CPS
12 SEC ON
40 SEC OFF
12 SEC ON
40 SEC OFF
12 SEC ON
60 SEC OFF
12 SEC ON
80 SEC OFF
12 SEC ON
100 SEC OFF
12 SEC ON
120 SEC OFF
200
400
600
800
1000
1200
1400
1600
1800
2000
2200
2400
2600
3000
MISSION TIME IN SECONDS
THERMAL MISSION DUTY CYCLE
10-50 MS PULSES
@1CPS
100 SEC ON
95 SEC ON
10-170 MS PULSES
@ 1 CPS
10-50 MS PULSES
@ 1 CPS
8 MIN.
HEAT SOAK
30 MIN.
HEAT SOAK
100
200
300
400
500
600
700
1100
1200
1300
1400
3100
3200
3300
3400
3500
3600
MISSION TIME IN SECONDS
WORST CASE DUTY CYCLE
10 SECOND BURNS
3 MIN. OFF
4 MIN. OFF
4 MIN OFF
10-50 MS PULSES
@ 5 CPS
@ 2 CPS
@ 1 CPS
10 SEC BURN
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
30
32
MISSION TIME IN MINUTES
THERMAL DUTY CYCLE

FIGURE 4.3-2

Figure 4.3-3 Engine S/N RDV-2B in Altitude Chamber

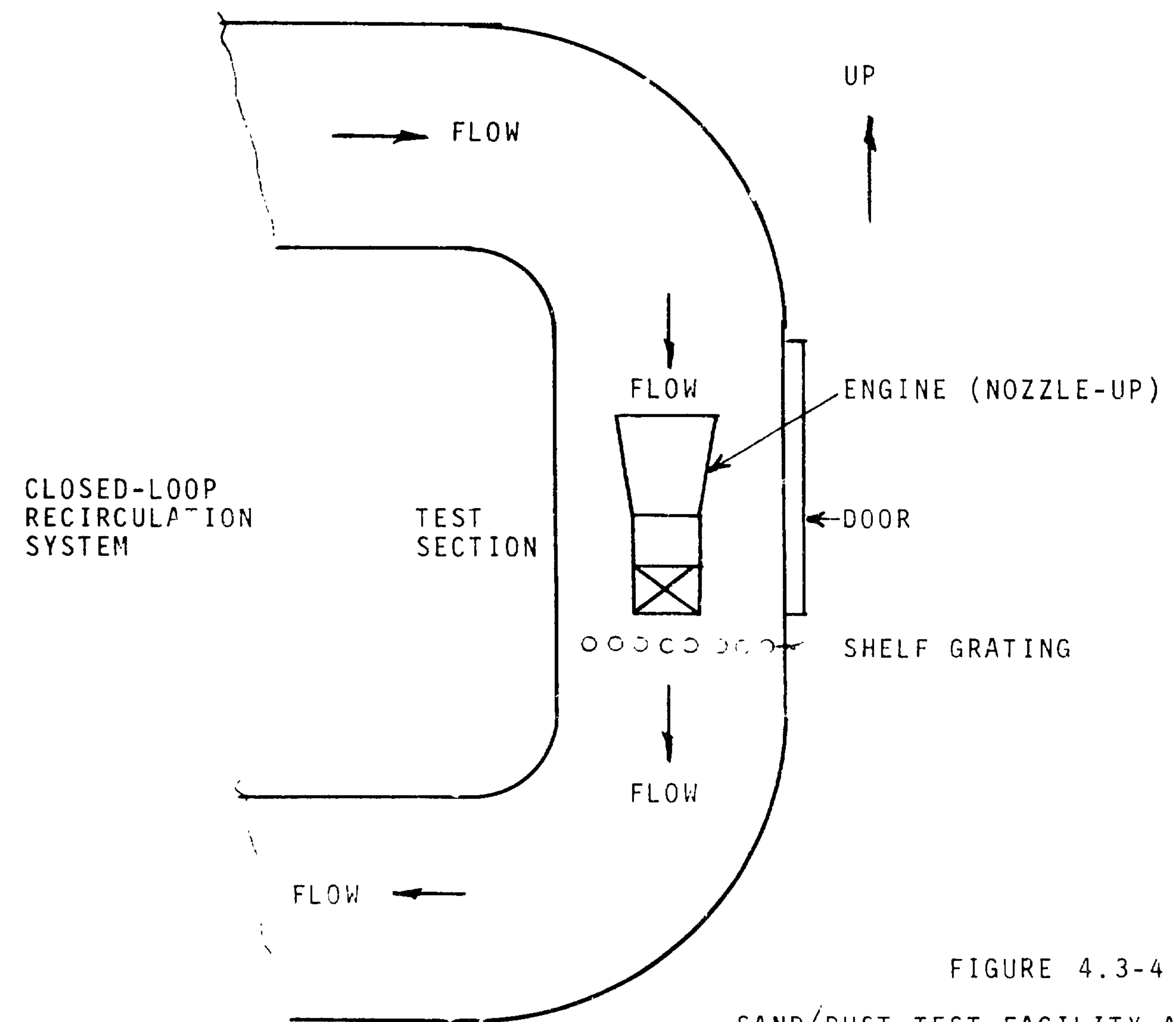


FIGURE 4.3-4

SAND/DUST TEST FACILITY AND SET-UP

Figure 4.3-5 Model 8701 RCS Engine S/N FT-2A

Figure 4.3-6 Cycle 1 - Post Salt Spray June 25, 1973

Figure 4.3-7 Cycle 1 - Post Sand and Dust Nozzle Up Attitude
June 25, 1973

Figure 4.3-8 Cycle 1 - Post Sand and Dust Nozzle Up Attitude
June 25, 1973

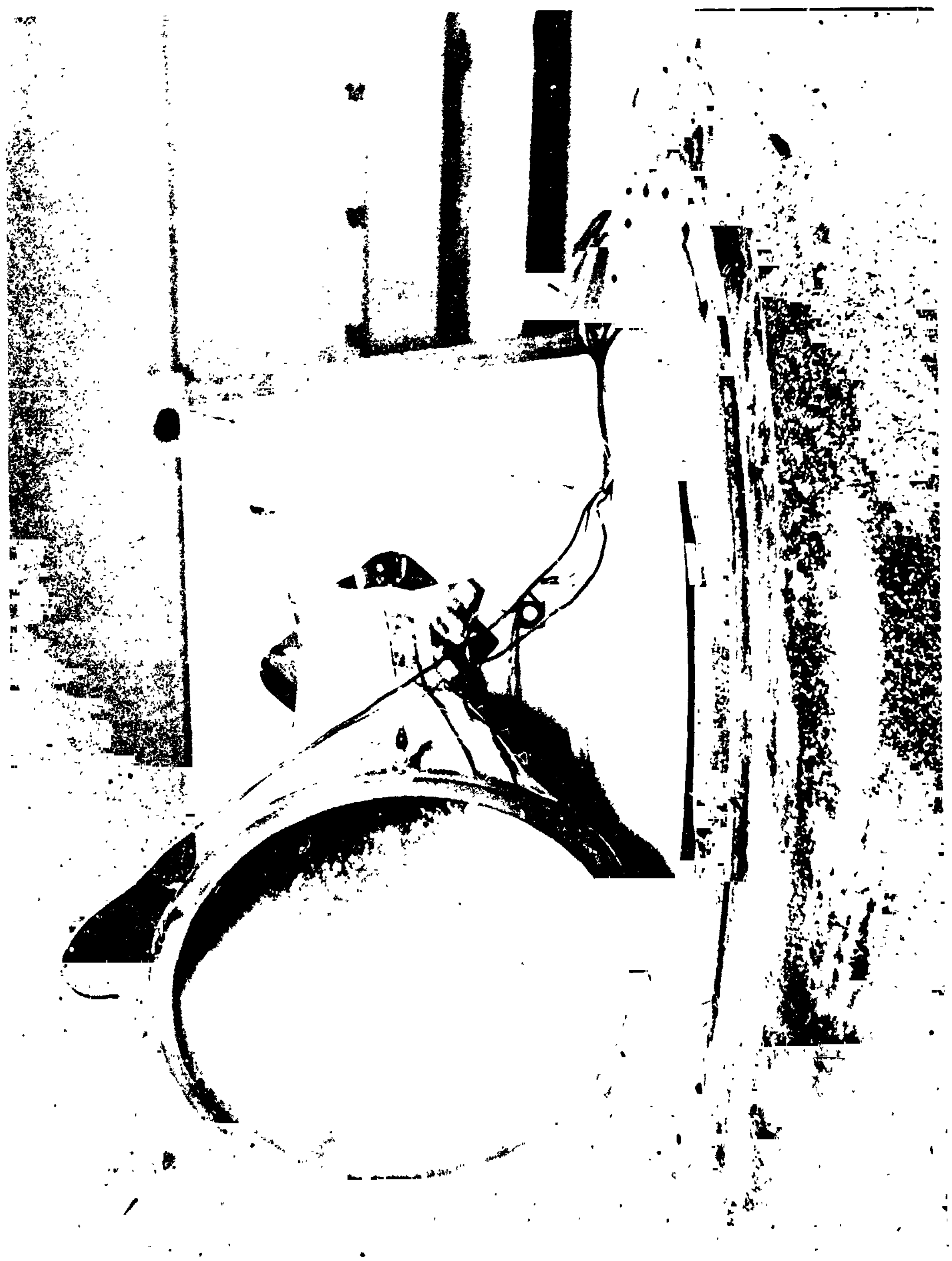


Figure 4.3-9 Cycle 1 - Sinusoidal Vibration June 26-27, 1973

Figure 4.3-10 Cycle 1 - Sinusoidal Vibration June 26-27, 1973

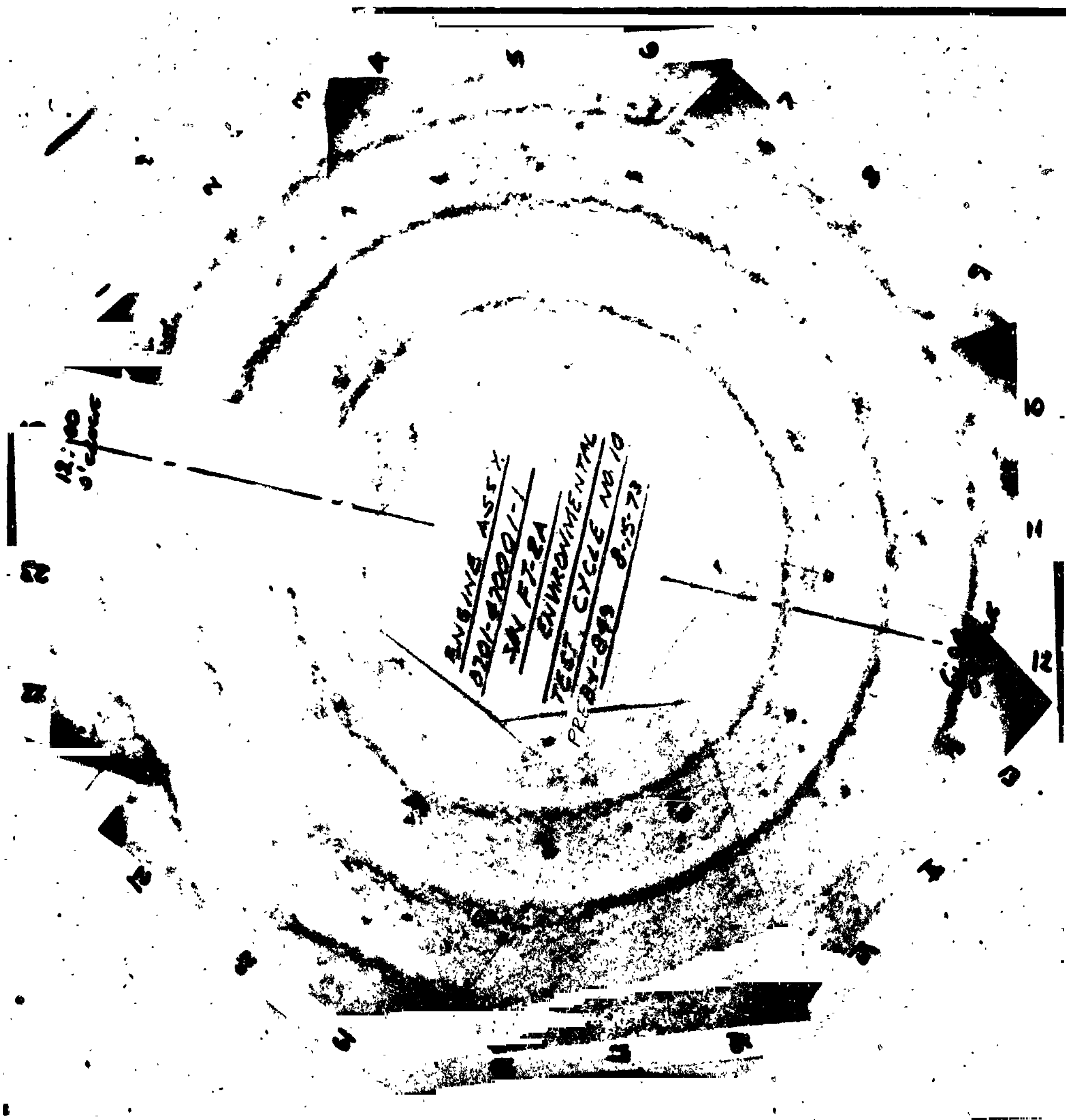


Figure 4.3-11 Pre Hot Fire - 10th Environmental Cycle

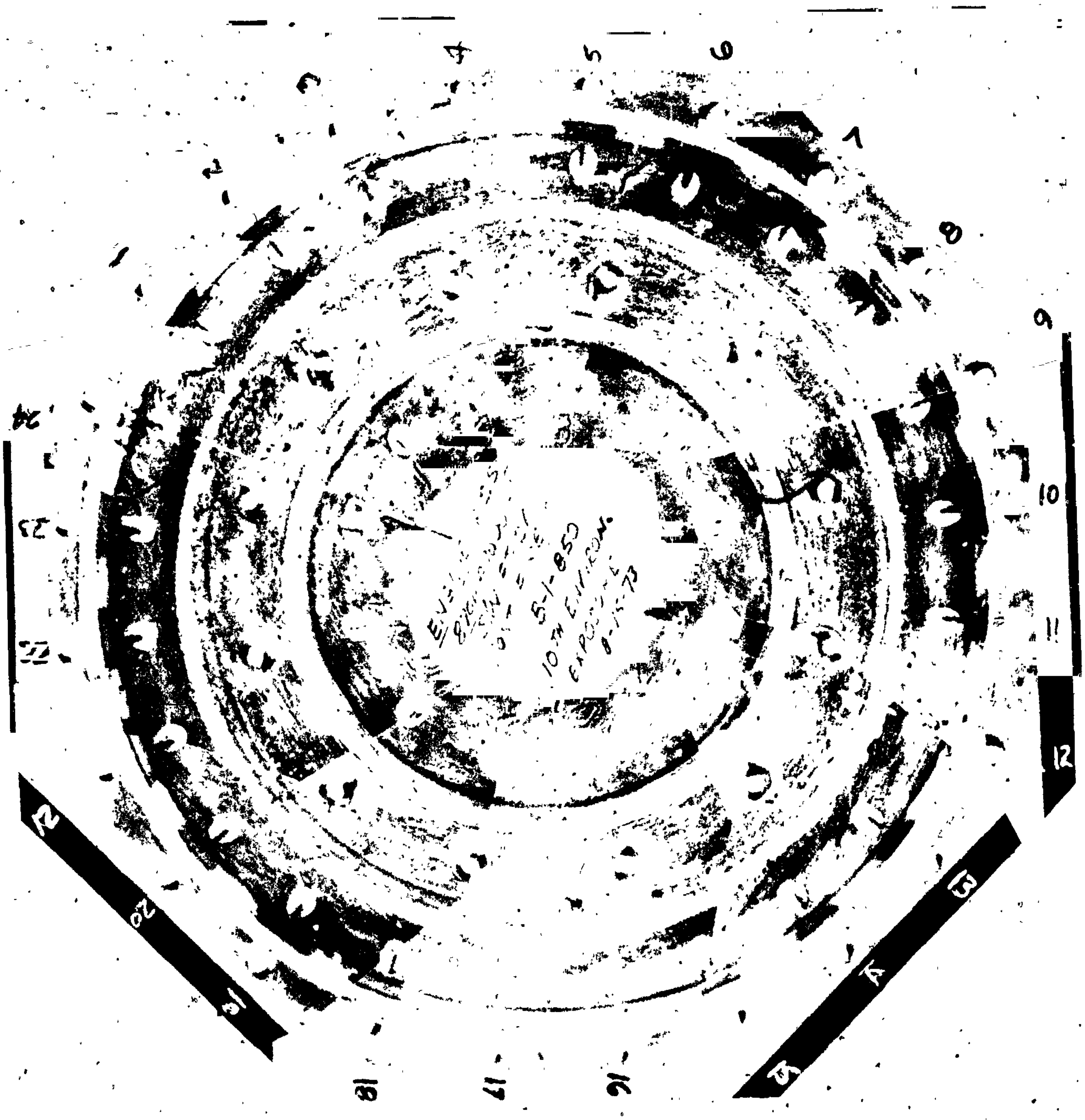


Figure 4.3-12 Post B-1-850 - Engine S/N FT-2A
Post Environmental Test - $\epsilon = 40/1$

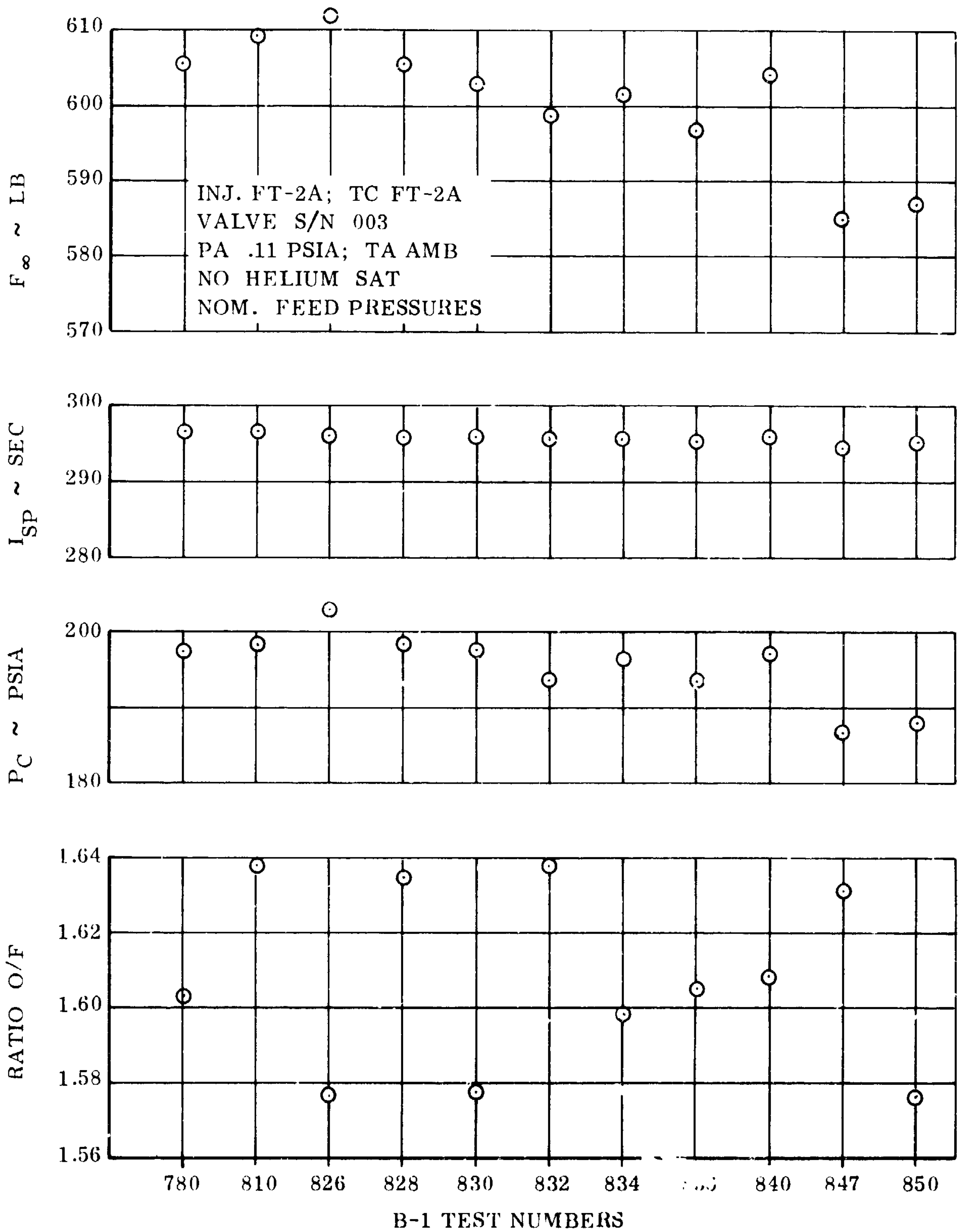
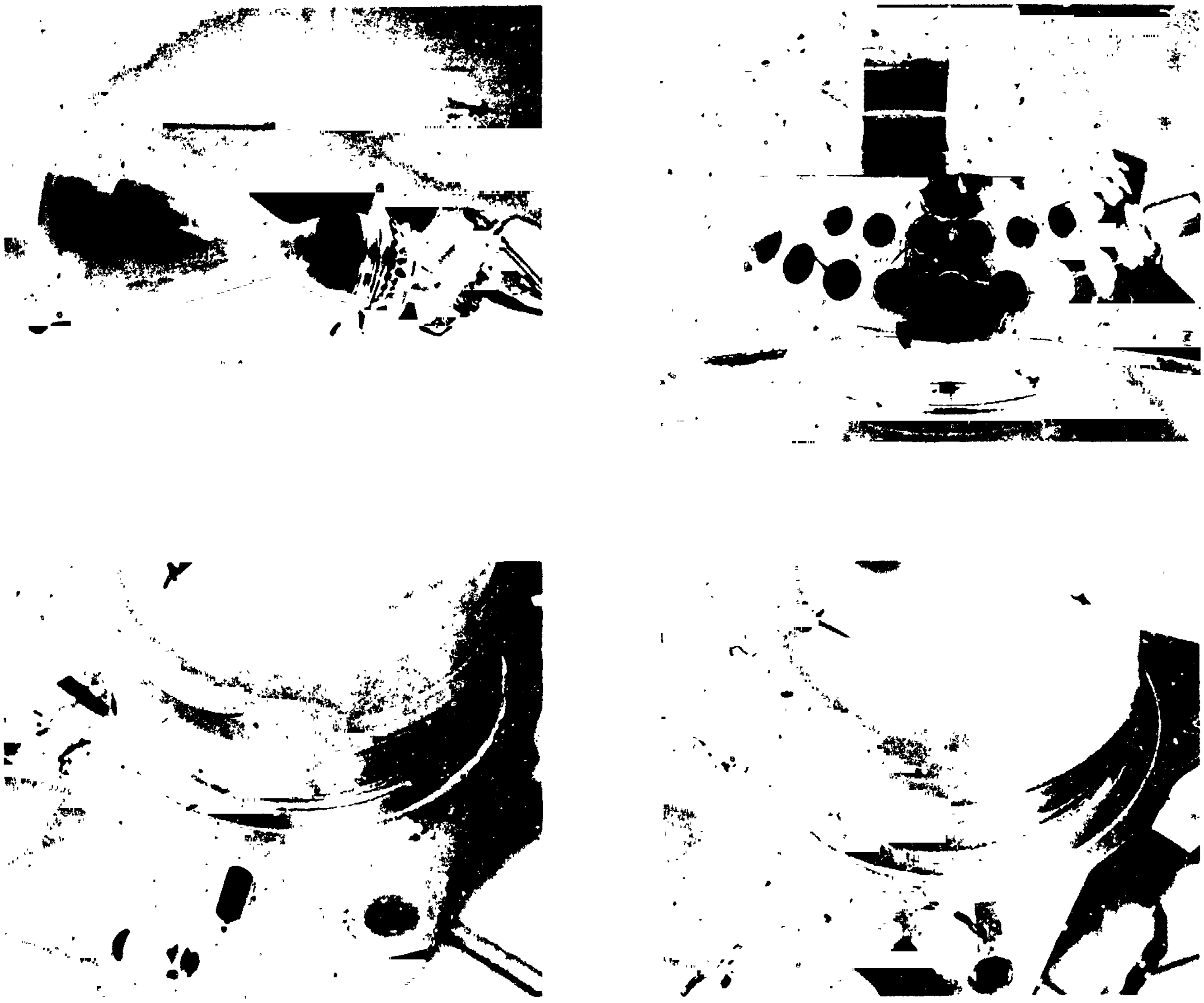


Figure 4.3-13 Environmental Tests Performance



Nominal Input: $1.6\ g^2/Hz$
PSD Output Response: $130\ g^2/Hz$
Amplification Factor: 9
Maximum Amplitude: 303 g's (3σ)

Figure 4.3-14 Engine S/N FT-2A Post Random Vibration

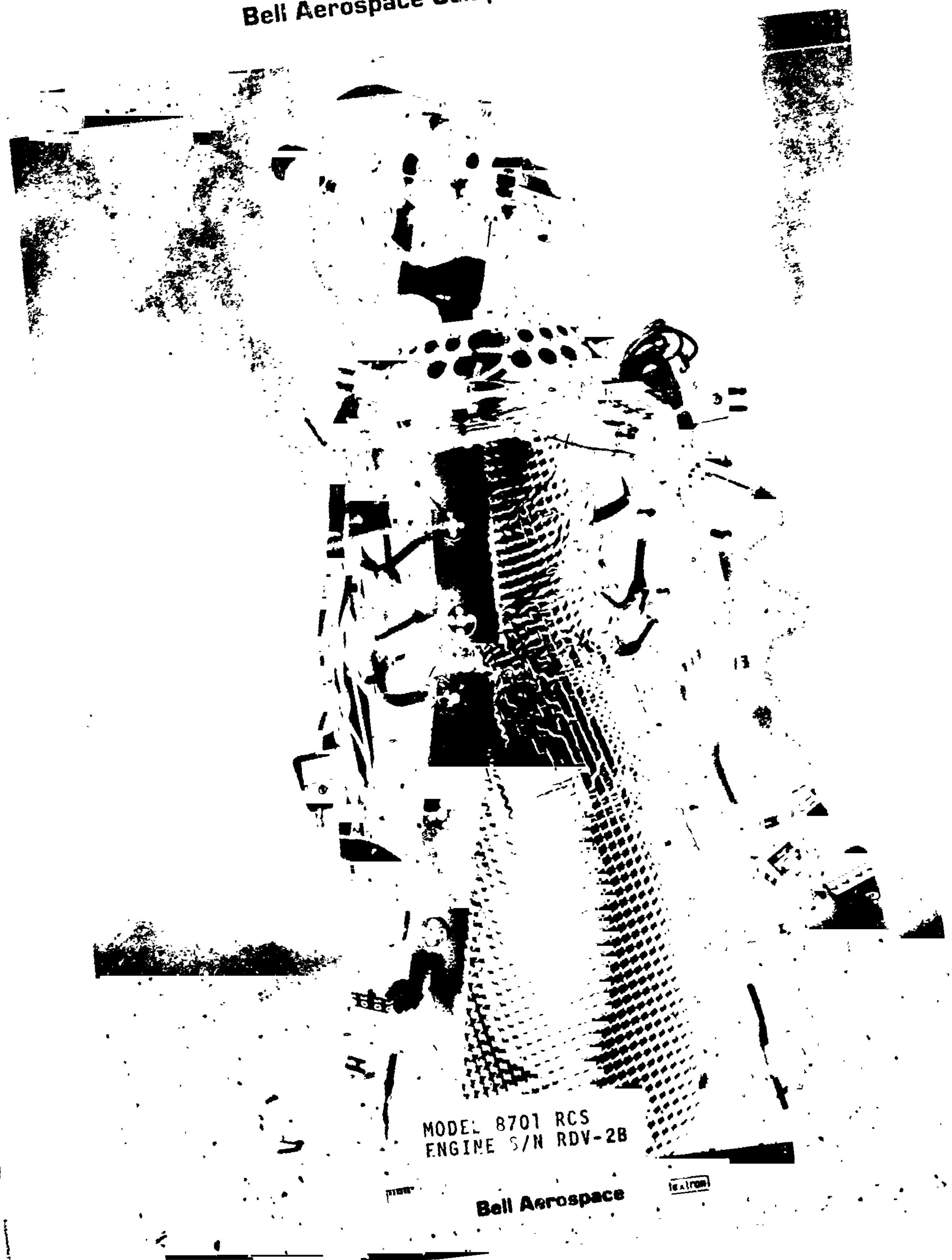


Figure 4.3-15 Model 8701 RCS Engine S/N RDV-2B

Engine S/N RDV-2B-1
e = 35
Test B-1-807
Nominal Feed Conditions

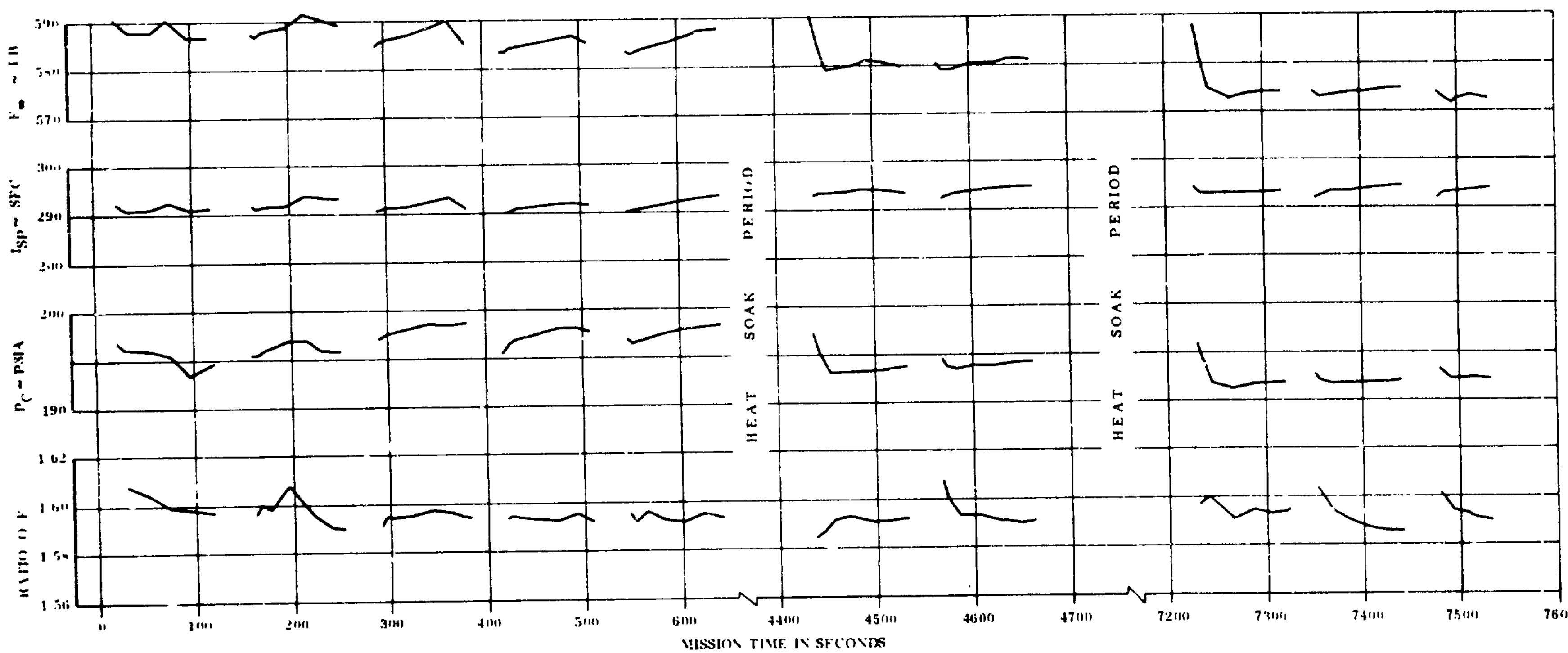


Figure 4.3-16 Worst Case Mission Performance, Engine RDV-2B

Engine S/N RDV-2B-1
$\epsilon = 35$
Test B-1-848
Nominal Feed Conditions

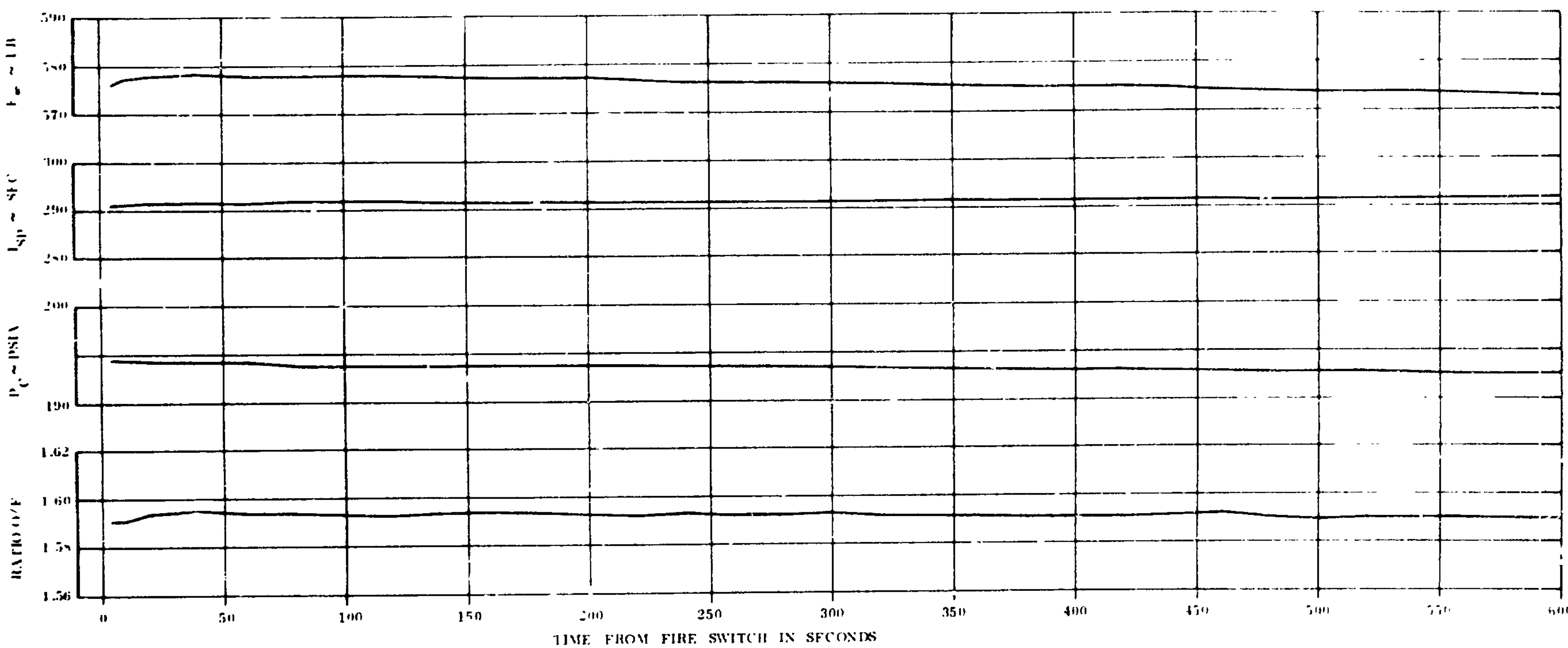


Figure 4.3-17 Endurance Test Performance, Engine RDV-2B

(BASED ON ENGINE S/N RDV-2B WITH PROPELLANT METERING SYSTEM) OT = EPW = 50 MS, f = 1 CPS, P_c = 200 PSIA, O/F = 1.6, ϵ = 40

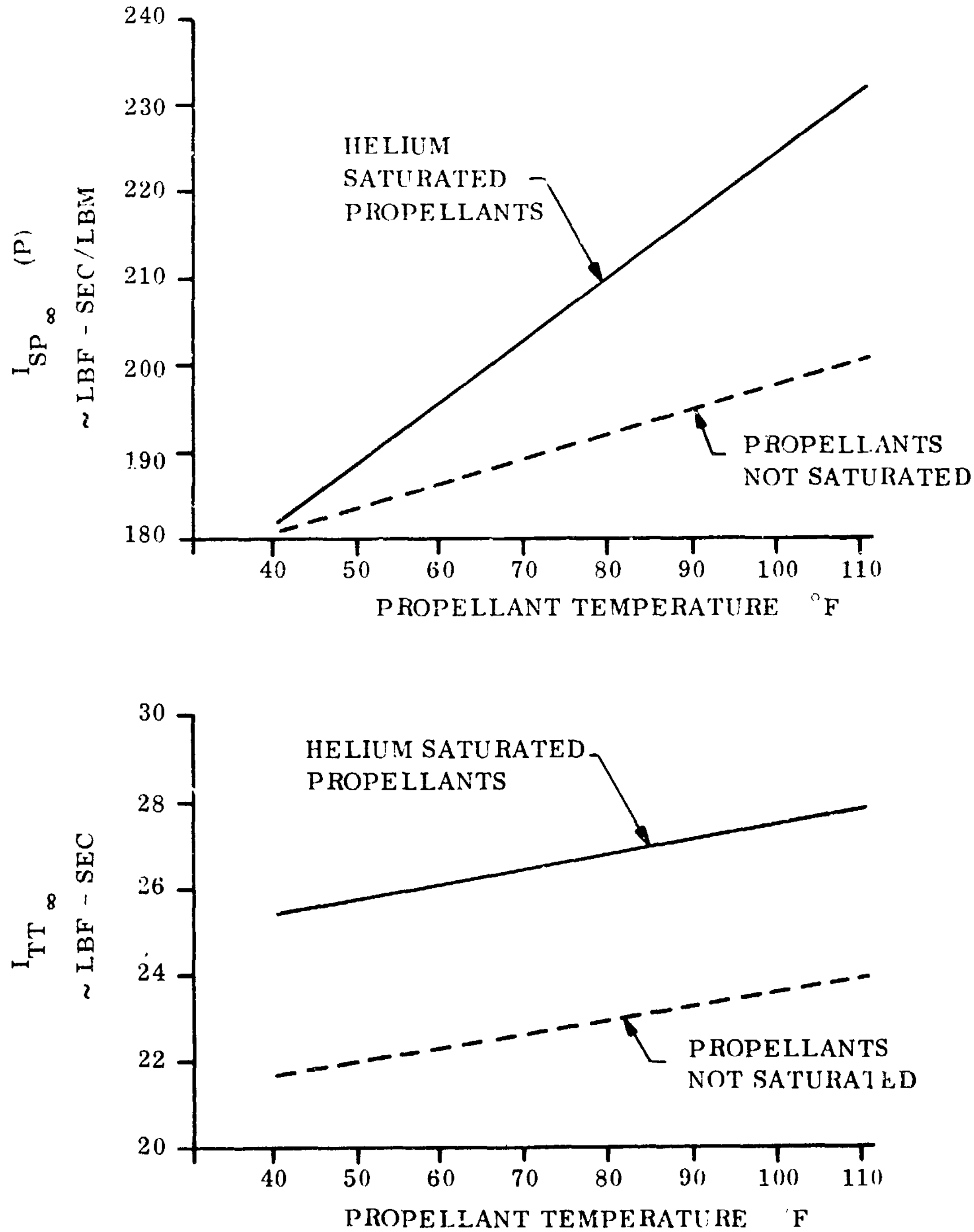
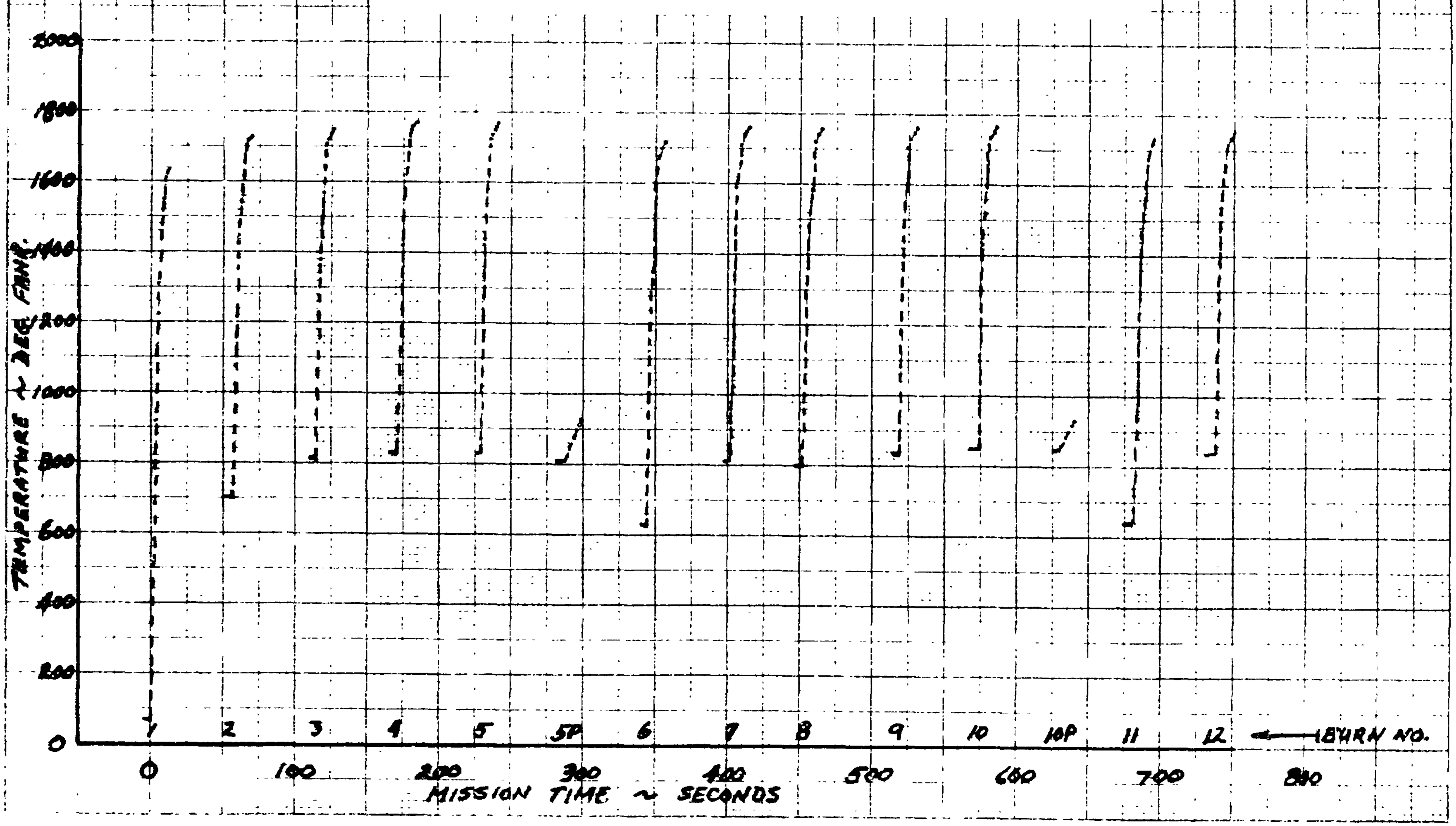


Figure 4.3-18 Pulsing Performance vs Propellant Temperature

FIGURE 4.3-19 TYPICAL THERMAL MISSION DUTY CYCLE THROAT TEMPERATURE PROFILE

TEST B-1-930



VARIATION OF STEADY STATE PERFORMANCE
WITH OPERATING CONDITION
(ENGINE S/N RDV-2B)

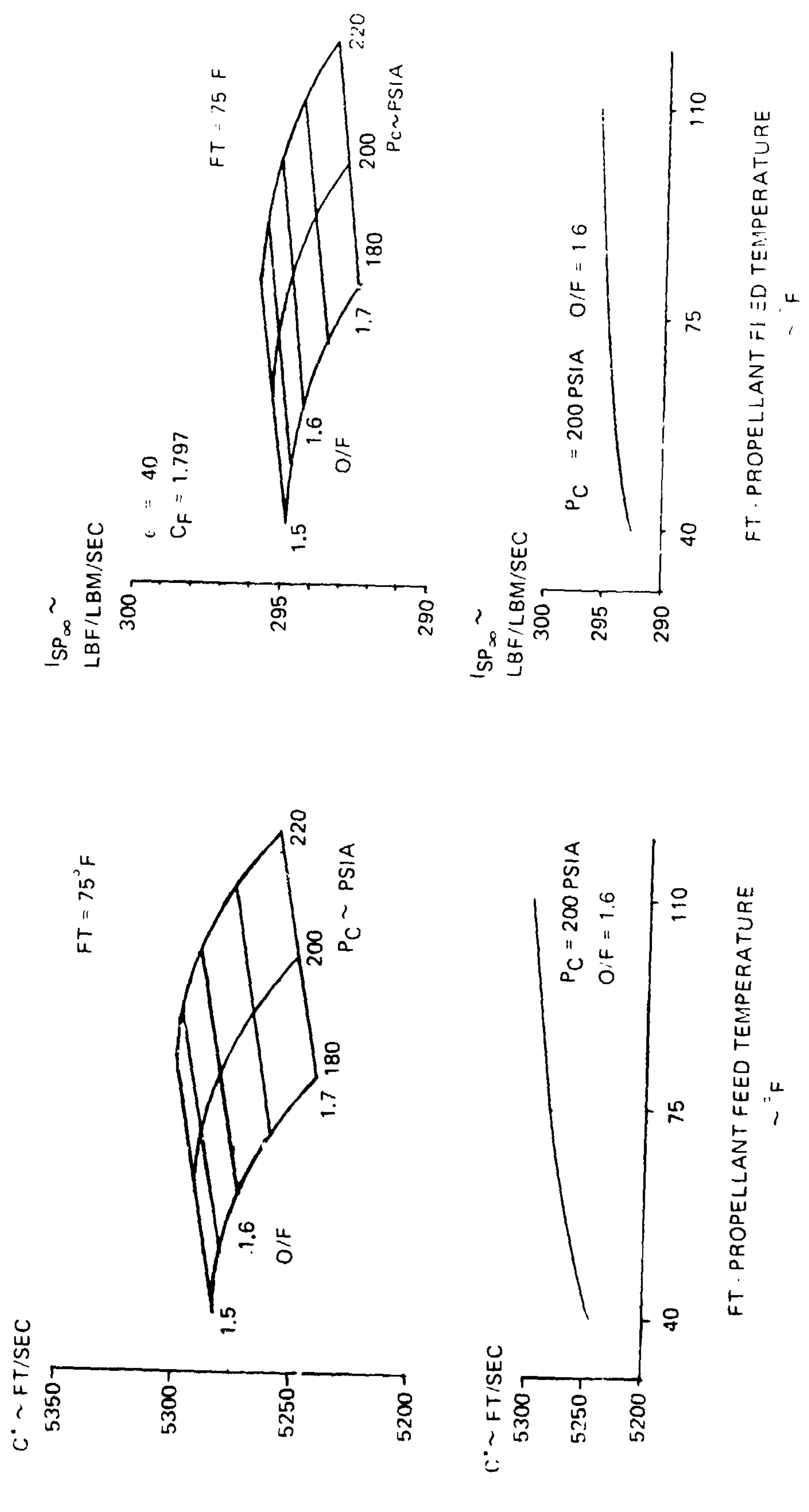


FIGURE 4.3-20

○ ENGINE S/N RDV-2B, UNSATURATED PROPELLANTS

● ENGINE S/N RDV-2B, HELIUM SATURATED PROPELLANTS

◇ ENGINE S/N FT-2A, UNSATURATED PROPELLANTS

$\epsilon = 40$, $P_c = 200$ PSIA, O/F = 1.6, $\overline{FT} = 75°F$, PROPELLANT METERING SYSTEM, f = 1 CPS

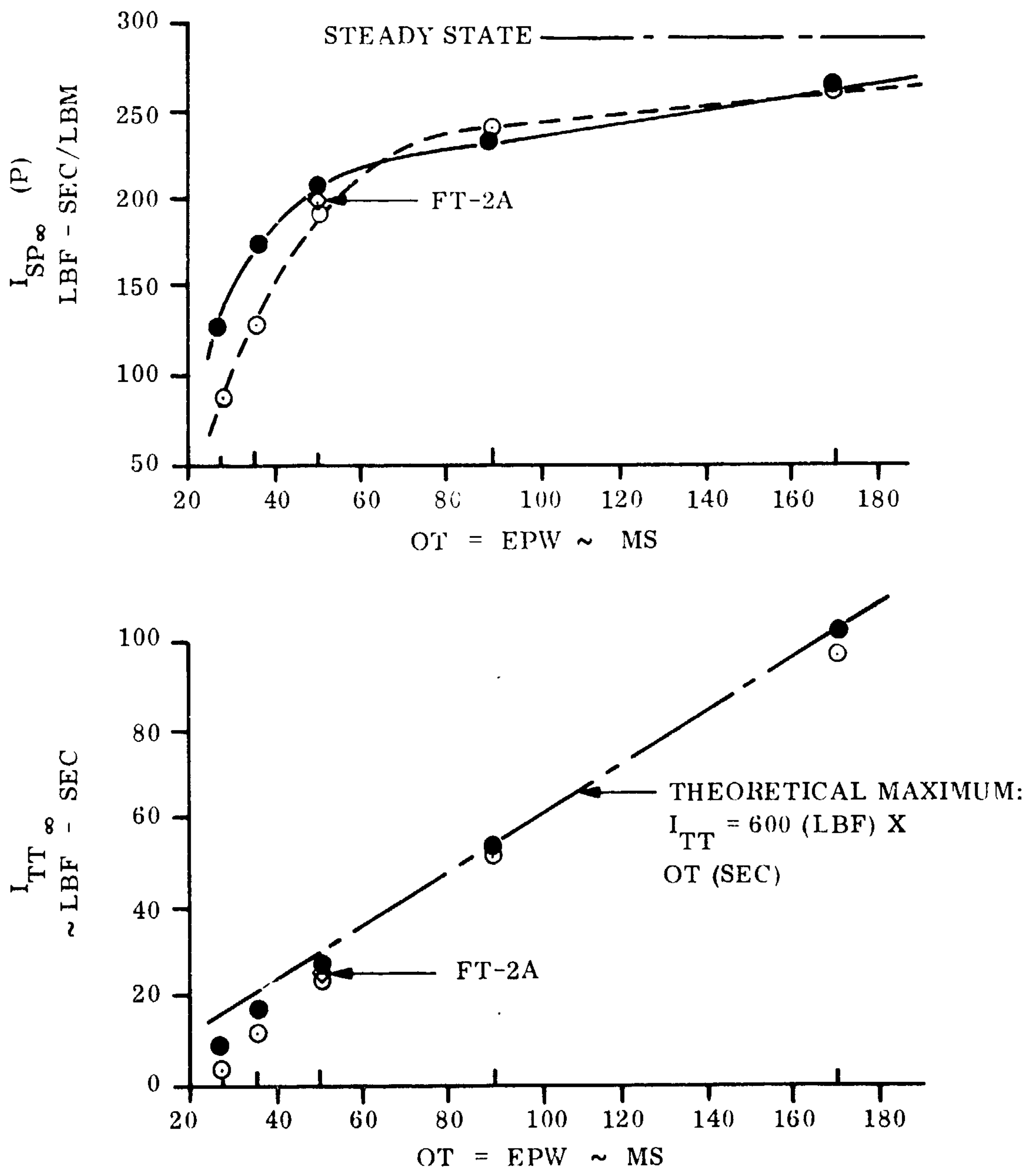


Figure 4.3-21 Pulsing Performance vs Pulse Width

⊙ ENGINE S/N RDV-2B. UNSATURATED PROPELLANTS

● ENGINE S/N RDV-2B, HELIUM SATURATED PROPELLANTS

◇ ENGINE S/N FT-2A, UNSATURATED PROPELLANTS

ϵ = 40, P_c = 200, O/F = 1.6, $\overline{FT}$ = 75°F, PROPELLANT METERING SYSTEM, OT = EPW = 50 MS

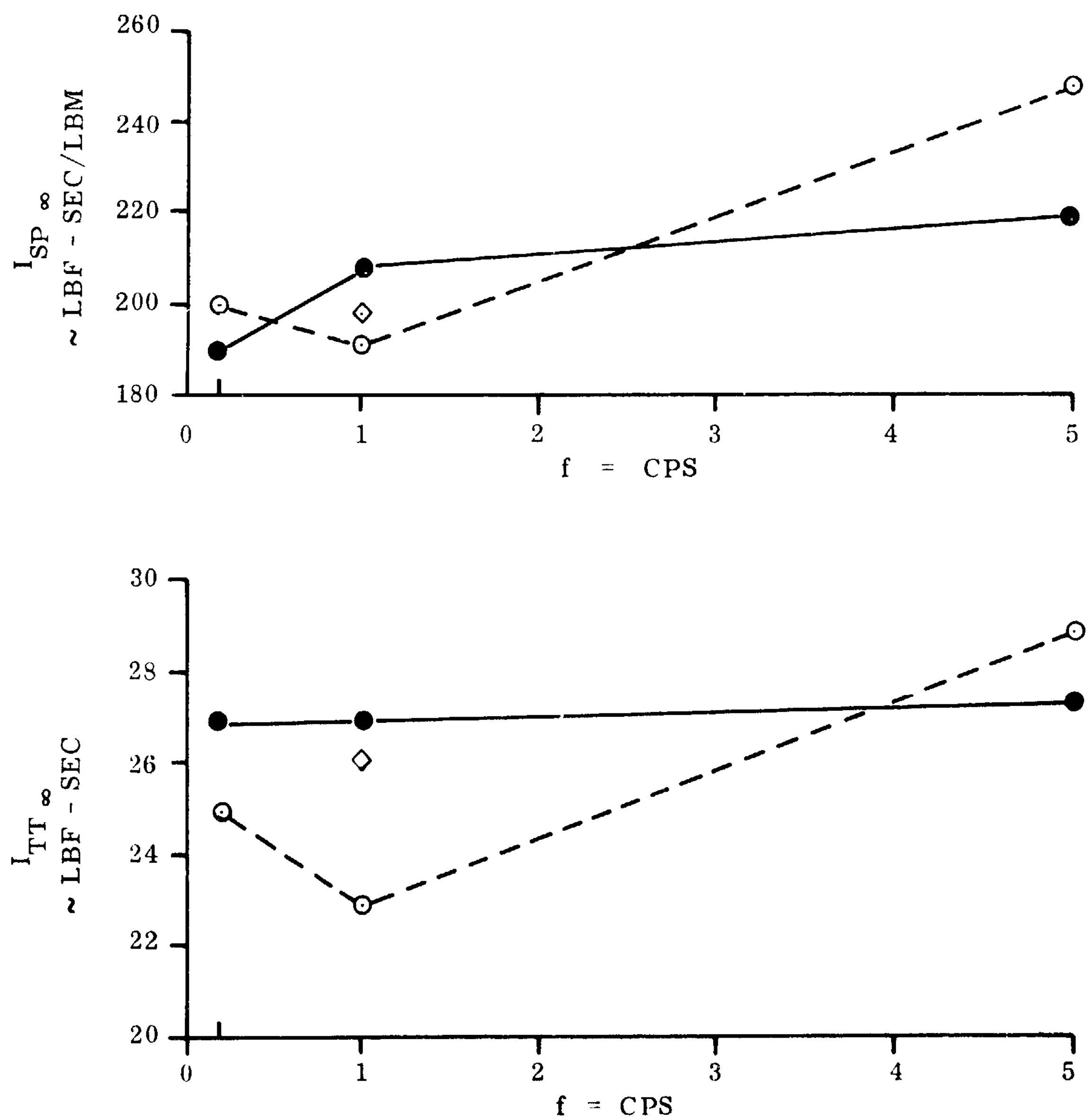


Figure 4.3-22 Pulsing Performance vs Pulsing Frequency

PULSING PERFORMANCE VERSUS STEADY STATE OPERATING CONDITION

(BASED ON ENGINE S/N RDV-2B WITH PROPELLANT METERING SYSTEM)

OT = EPW = 50 ms, f = 1 CPS, FT = 75°F, ε = 40

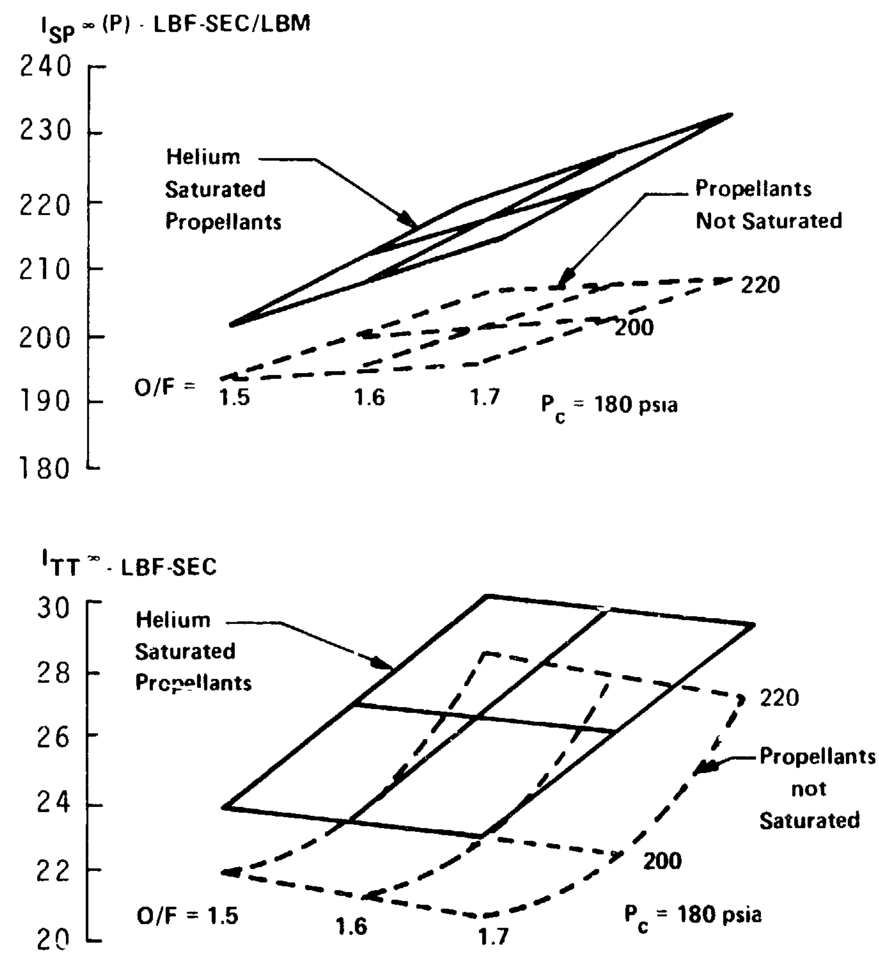


FIGURE 4.3-23

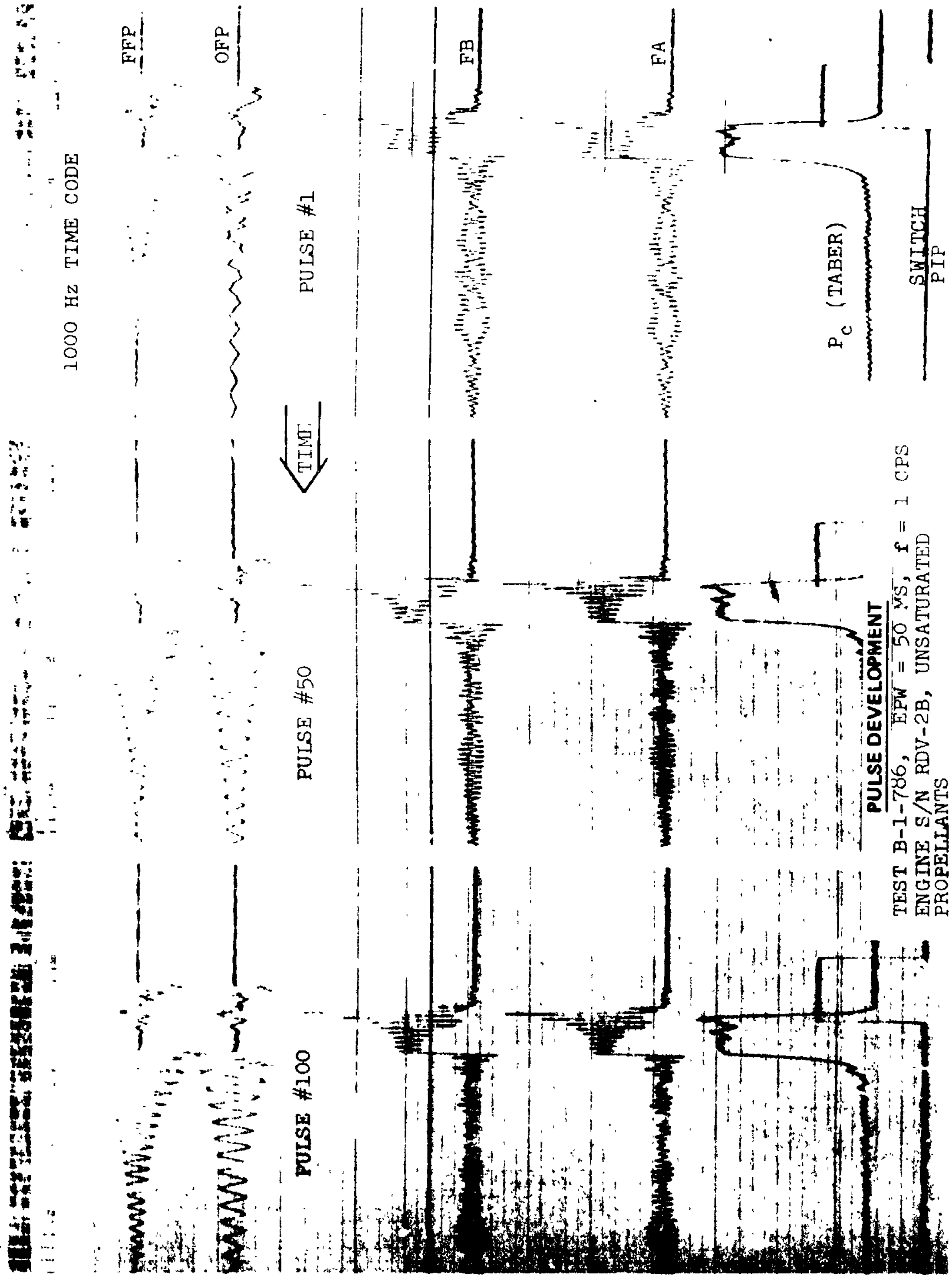

1000 Hz TIME CODE
FFP
OFP
PULSE #1
PULSE #50
PULSE #100
TIME
FB
FA
P_c (TABER)
SWITCH PIP
PULSE DEVELOPMENT
TEST B-1-786, EPW = 50 MS, f = 1 CPS
ENGINE S/N RDV-2B, UNSATURATED
PROPELLANTS

FIGURE 4.3-24

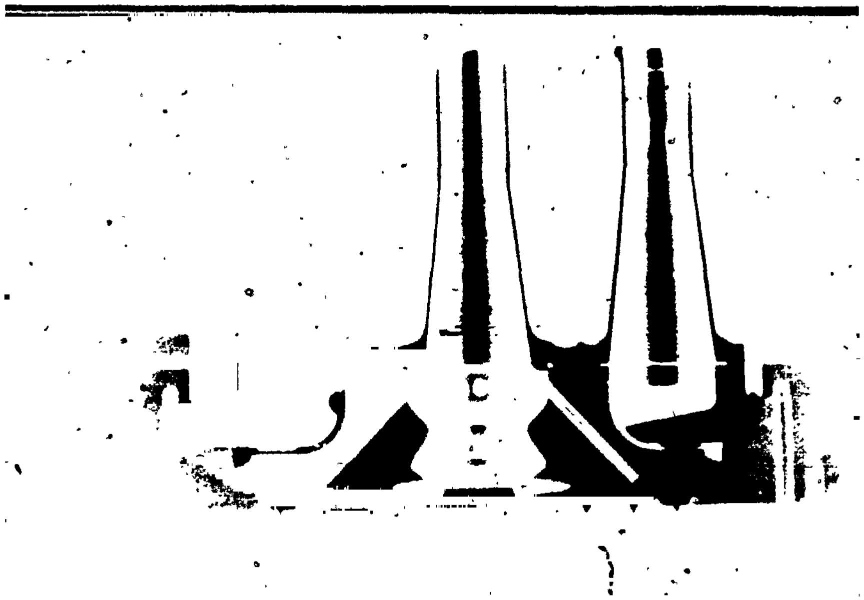
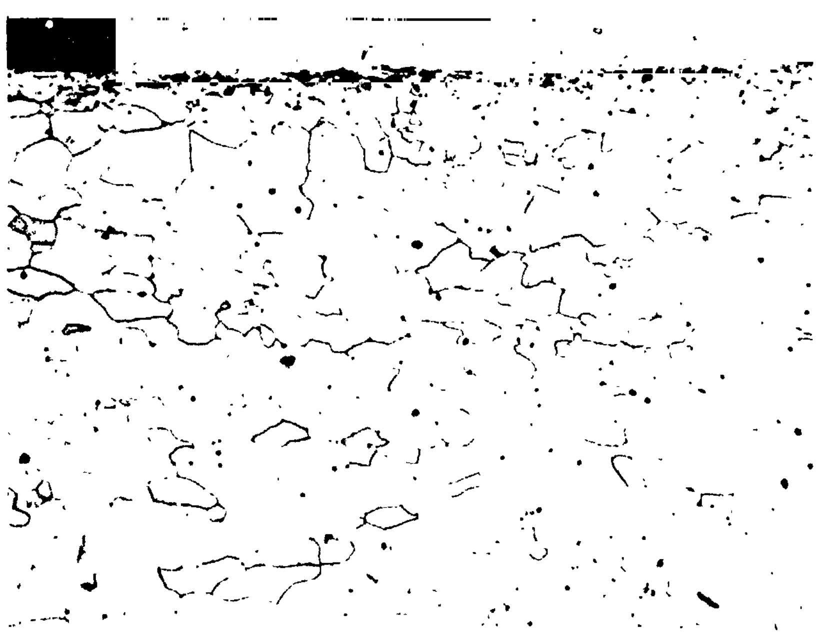


Figure 4.3-25 Cross Section of Injector From Engine FT-2A Component Location and EB Welds are Visible



MAG: 100X
MATERIAL: Cb-1%Zr

CENTRAL REGION OF INJECTOR FACE

Figure 4.3-26 Microstructure and Surface Profiles of Injector Face From FT-2A Engine

Figure 4.3-27 Segment of Injector Face From Engine RDV-2 Metallurgically Evaluated.

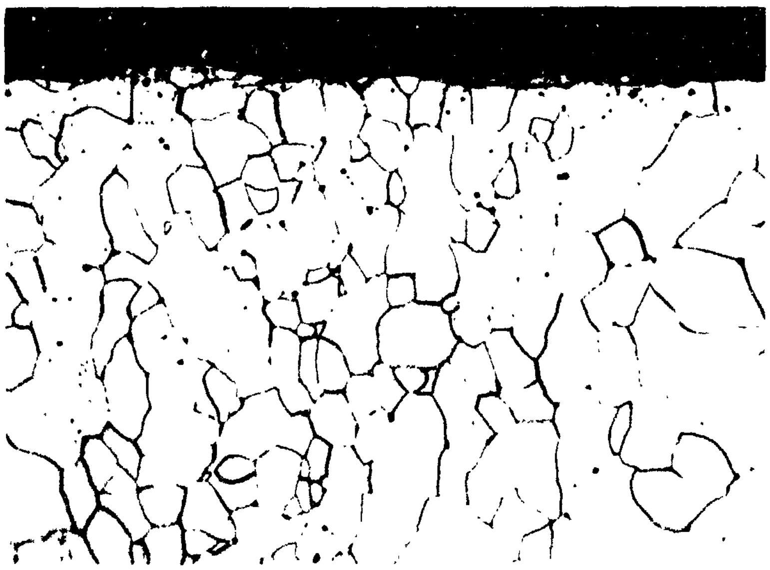


Figure 4.3-28 SN RDV-2B Microstructure of Central Region of Injector Face. Mag: 100X

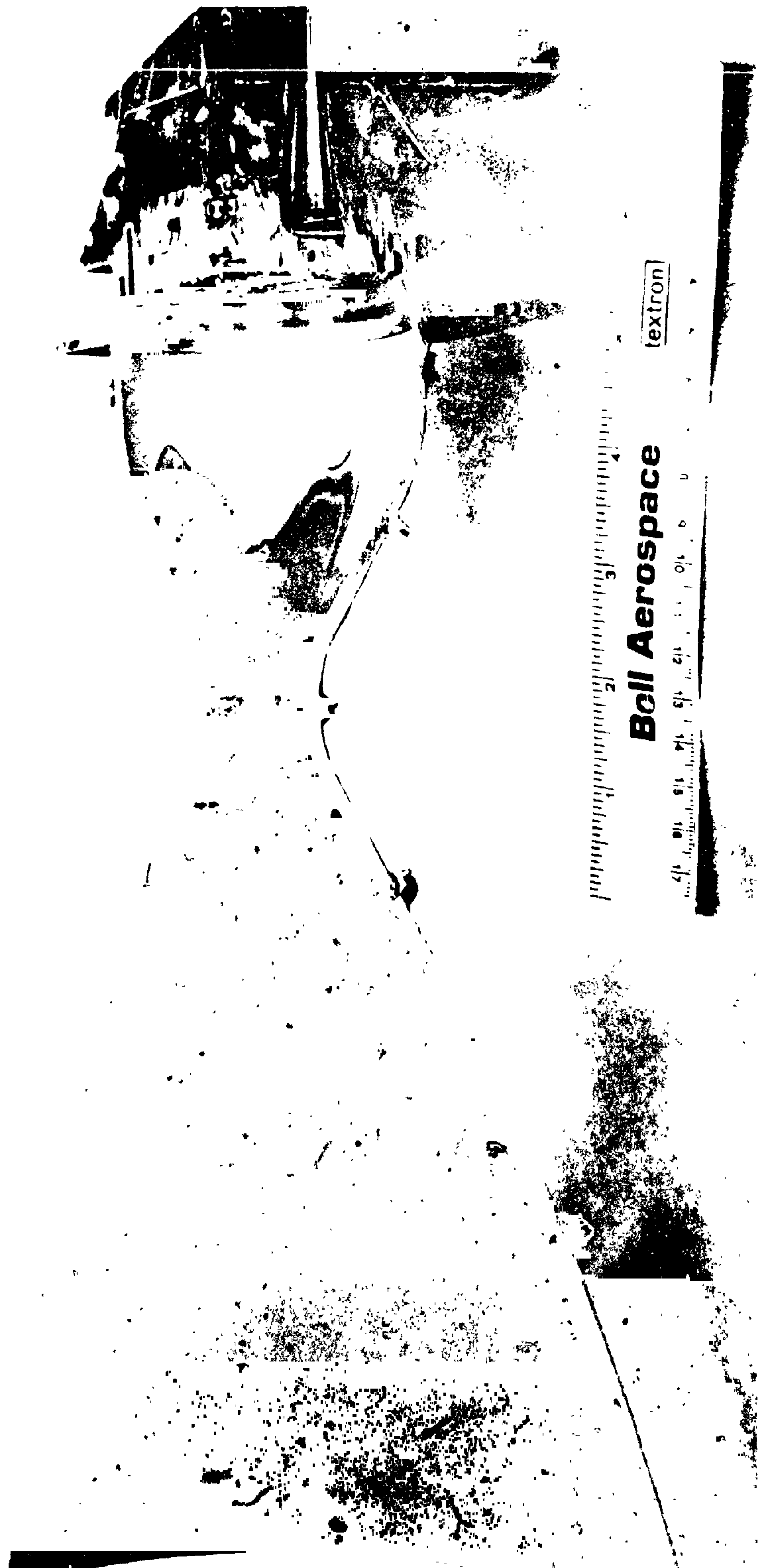
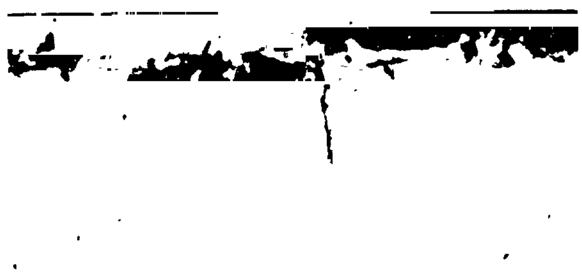
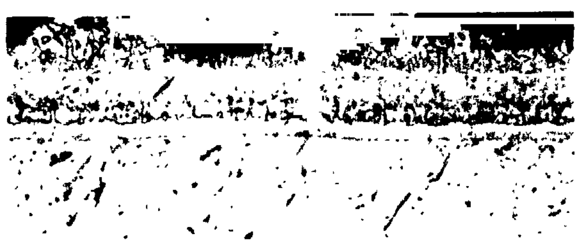


Figure 4.3-29 Overall View of Segment Removed From RDV-2B and Evaluated by Metallurgical Laboratory. Note Cool Chamber Area, Near Injector



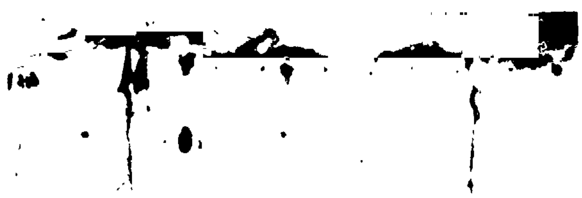
LOCATION A. CHAMBER (CLOSE TO INJECTOR)

LOCATION B. CHAMBER

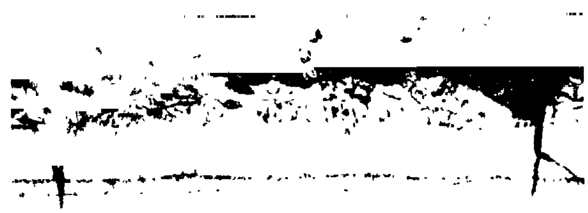


LOCATION C. CONVERGENT PORTION OF THRUST CHAMBER

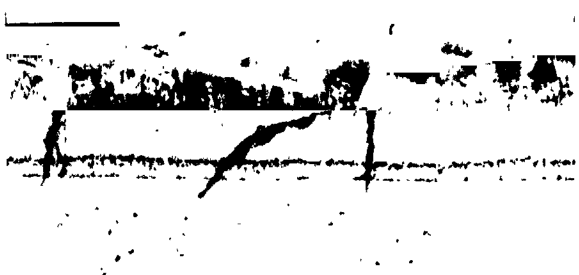
LOCATION D. THROAT OF THRUST CHAMBER



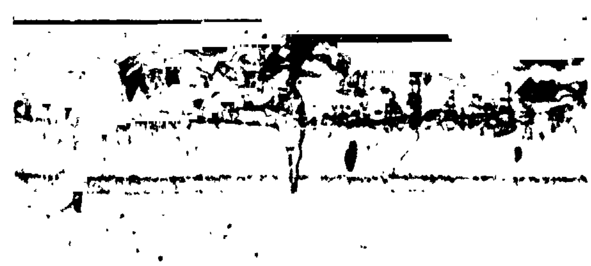
LOCATION E. DIVERGENT PORTION OF THRUST CHAMBER



LOCATION F. START OF EXTENSION NOZZLE



LOCATION G. MID-SECTION NOZZLE



LOCATION H. END OF NOZZLE

ALL VIEWS MAG: 200X

Figure 4.3-30 Section Views Showing Structure of R512E Coating On Interior of Thrust Chamber in Engine RDV-2B

MAG: 200X

Figure 4.3-31 SCb 291 Coated With R512E as Received

APPENDIX I

FUEL VORTEX COOLING

APPENDIX I

FUEL VORTEX COOLING

Bell Aerospace Company has devoted several years of effort to methods of rocket engine gas side film cooling with N_2O_4 and amine fuels. The objective of the investigations was to establish design techniques which would provide a uniform wall thermal environment at predictable values with minimum engine performance loss. Various types of fuel and oxidizer film cooling were evaluated primarily by test firings at Bell, and fuel vortex film cooling was determined to provide the desired characteristics (Ref. 1, 2). Fuel vortex film cooling is provided by tangential injection of fuel near the chamber wall and the development of a fuel vortex film or barrier at the head end of the chamber.

Most of the testing was conducted with 3.4-inch diameter hardware at an L* of 14 inches at a chamber pressure of 200 psia. Radiation-cooled, coated columbium thrust chambers were used with pyroscanners to obtain continuous thermal profiles of the engine during test. This allowed a direct measurement of temperature uniformity and maximum wall temperature from the transient through equilibrium. The temperature data allowed computation of the effective gas side heat transfer coefficients and the driving gas temperature with the assumption of one-dimensional radial heat flow. The driving temperature is essentially equal to the wall temperature for a completely insulated thrust chamber.

The maximum wall temperature which occurs near the throat is a function of the ratio of barrier flow to total propellant flow (ρ) for a given configuration. A statistical multi-regression correlation analysis for the 600 lb-engine thermal data was employed to assess the significance of the various operating parameters on the maximum steady-state wall temperature.

$$T_{max} = A\rho + B(r_o) + C(P_c) + N$$

where T_{max} is maximum steady-state wall temperature, r_o is overall mixture ratio, ρ is the ratio of barrier flow to total propellant flow and A, B, C, N are constants. It was found that over the range of parameters investigated (that is, $0.08 < \rho < 0.14$, $125 < P_c < 250$ psia, $1.4 < r_o < 1.8$), ρ is the only significant parameter and the data for a given geometric configuration are correlated well by the relationship,

$$\rho = \frac{N^1 - T_{max}}{A}$$

where N^1 is an averaged constant. Figure I-1 is a plot of T_{max} versus ρ for tests with ρ above 9 percent. Good agreement is indicated.

An analytical model was developed to incorporate the effects of combustor geometry and thrust on the relationship between T_{max} and ρ. The basic hypothesis underlying the analytical model is that the ratio of film coolant flow rate and the heat rejected to the barrier film, as it moves from the point of injection to the throat, is a constant. The development of a fuel vortex cooling parameter ρ^* from that hypothesis is presented below. The comparison of test results is shown in Figure I-2. These results include data from flight-type and non-flight type engines. At barrier flows below 13%, the flight type engine data indicates somewhat higher temperatures than non-flight type engines. The higher thrust data is associated with the work at Bell for the Space Shuttle Orbit Maneuvering Engine (OME) under Contract NAS 9-12803. The data show that the requirement of cooling concept scalability has been demonstrated.

FIGURE I-1.
Temperature-Barrier Flow Correlation

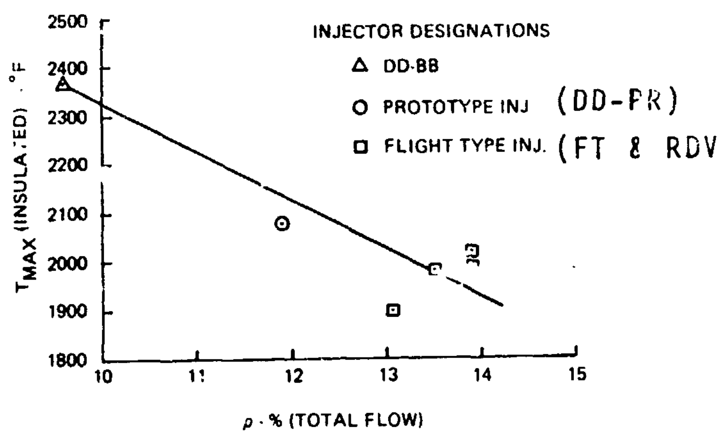
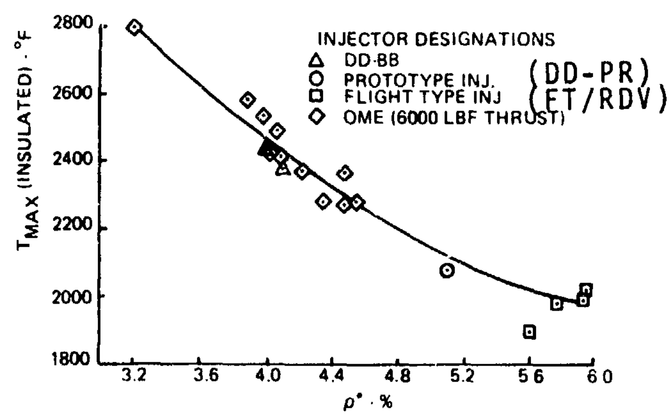


FIGURE I-2.
Vortex Cooling Parameter Correlation



DERIVATION OF VORTEX COOLING PARAMETER, ρ^*

HYPOTHESIS $\dfrac{\dot{\omega}_{c_1}}{\dot{\omega}_{c_2}} = \dfrac{\dot{Q}_1}{\dot{Q}_2}$ — RATIO OF FILM COOLANT FLOW RATE, $\dot{\omega}_c$, TO TOTAL HEAT REJECTED TO FILM, $\dot{Q}$ IS CONSTANT

$$\dot{Q} = h_c \pi D_c L \overline{\Delta T}$$

WHERE
- h_c = HEAT TRANS COEFFICIENT CORE TO FILM
- D = DIAMETER OF CHAMBER
- L = ENGINE LENGTH VORTEX LIP TO THROAT
- $\overline{\Delta T}$ = AVERAGE TEMPERATURE GRADIENT CORE TO FILM

DEFINE $\rho = \dfrac{\dot{\omega}_c}{\dot{\omega}_t}$ WHERE $\dot{\omega}_t$ = TOTAL PROPELLANT FLOW RATE

SUBSTITUTING

$$h_c = \frac{C}{D^{1.8}} \left(\frac{4\dot{\omega}_t}{\pi\mu}\right)^{0.8} P_r^{0.4}$$

$$\dot{\omega}_t = \frac{g P_c A_t}{c^*}$$

- μ = KINEMATIC VISCOSITY
- P_r = PRANDTL NUMBER
- A_t = AREA OF THROAT
- c^* = CHARACTERISTIC VELOCITY

AND ASSUMING $\overline{\Delta T}_1 = \overline{\Delta T}_2,\ P_{r_1} = P_{r_2},\ \mu_1 = \mu_2$

$$\frac{\rho_1}{\rho_2} = \frac{\dfrac{L_1}{D_1^{\,0.8}}\left(\dfrac{c_1^*}{P_{c_1} A_{t_1}}\right)^{0.2}}{\dfrac{L_2}{D_2^{\,0.8}}\left(\dfrac{c_2^*}{P_{c_2} A_{t_2}}\right)^{0.2}}$$

DEFINE

$$\rho^* = \frac{\rho}{\dfrac{L}{D^{0.8}}\left(\dfrac{c^*}{P_c A_t}\right)^{0.2}}$$

$$\therefore \rho_1^* = \rho_2^*$$

AND WITH ρ_2^* ESTABLISHED BY TEST DATA FOR L_2 = 3.9, D = 3.4 IN. THRUST CHAMBER FOR N_2O_4/MMH PROPELLANTS.

THEN

$$T_{MAX} = 1833 + 3008 e^{-(15.2\rho^* + 590\rho^{*2})}$$

References

1 Berman, K., and Andrysiak, S.J., "Barrier Film Cooling Study," Journal of Spacecraft and Rockets, March 1972, Vol. 9, No. 3, p. 152.

2 Berman, K., Ferger, T., Roth, N., and Blessing, A.H., "Scaling of Performance and Thermal Environment in Fuel-Vortex-Cooled Rocket Engine," presented at AIAA/SAE 8th Propulsion Joint Specialist Conference, Nov. 29-Dec 1, 1972, New Orleans, La.

APPENDIX II

PHASE I TEST DATA

PART A

SUMMARY OF TEST DATA

INJECTOR S/N RDV-DD-PR-1

BARRIER % (ρ) = 13.1%

PERFORMANCE OF INJECTOR S/N RDV-DD-PR-1

Test Date	Test No.	Length of Run (Secs.)	P_c	r	C*	F_∞	$I_{sp\infty}$
8-22	B2-1388	2.1	223.9	1.537	5246	662.88	288.20
8-22	B2-1389	2.0	223.5	1.538	5239	661.47	287.81
8-22	B2-1390	2.0	201.2	1.512	5237	595.47	287.71
8-22	B2-1391	45.2	201.5	1.510	5250	597.34	288.35
			201.0	1.512	5267	600.21	289.41
			200.5	1.514	5264	602.11	290.26
			199.5	1.516	5271	600.81	289.65
8-22	B2-1392	1.9	200.0	1.274	5218	591.71	286.59
8-23	B2-1393	2.2	201.5	1.355	5245	596.76	288.11
8-23	B2-1394	45.3	201.0	1.357	5237	595.86	287.70
			200.2	1.358	5259	597.83	288.96
			199.7	1.356	5277	599.70	289.95
			198.6	1.359	5269	598.10	289.40
8-23	B2-1395	45.1	201.2	1.357	5226	596.46	287.10
			199.8	1.358	5249	596.63	288.43
			199.1	1 357	5264	596.63	288.43
			198.3	1.359	5262	597.90	289.28
						597.20	289.07
8-23	B2-1396	2.2	200.6	1.704	5175	594.10	284.23
8-23	B2-1397	45.1	201.8	1.738	5161	598.23	283.48
			201.3	1.737	5179	601.11	284.50
			200.9	1.739	5196	603.31	285.47
			199.4	1.740	5172	600.51	284.18
8-23	B2-1398	2.2	178.1	1.495	5194	527.46	285.42
8-23	B2-1399	2.2	177.6	1.606	5176	525.98	284.29
8-23	B2-1400	45.9	179.1	1.569	5200	530.94	285.71
			178.3	1.567	5223	532.43	286.87
			178.0	1.566	5241	534.54	287.92
			176.9	1.566	5225	532.75	287.03
8-23	B2-1401	2.2	224.8	1.485	5213	665.77	286.36
8-23	B2-1402	41.4	223.8	1.504	5216	663.45	286.53
			222.0	1.503	5232	662.92	287.35
			220.7	1.502	5228	662.77	287.21
8-23	B2-1403	2.2	200.5	1.494	5229	593.80	287.33
8-24	B2-1404	45.5	202.5	1.543	5224	600.31	286.96
			202.2	1.544	5265	603.80	288.16
			201.8	1.544	5265	606.01	289.20
			200.2	1.547	5241	602.92	287.90

STABILITY SERIES - INJECTOR S/N RDV-DD-PR-1

TEST DATE	TEST NO.	L.O.R.	P_c	r	DAMP TIME MS	BOMB SPIKE PRESS.	BOMB SIZE	HELIUM SAT.	$T_{PROP.}$	TIME OF BOMB SWITCH
8-29	D3-6413	1.5	198	1.57	1	613	2gr.	Yes	Amb.	.376
8-29	D3-6414	Biprop.	Valve	did not	open.	141	2gr.	Yes	Amb.	.389
8-29	D3-6415	Biprop.	Valve	did not	open.	86	2gr.	Yes	Amb.	.389
8-30	D3-6416	.5	189	1.22	1	510	2gr.	Yes	Amb.	.373
8-30	D3-6417	.5	195	1.15	1	597	2gr.	Yes	Amb.	.373
8-30	D3-6418	.5	182	1.52	No	Detonation		Yes	Amb.	.373
8-30	D3-6419	.5	182	1.60	1	528	2gr.	Yes	Amb.	.373
8-30	D3-6420	.5	222	1.62	1	660	2gr.	Yes	Amb.	.373
8-30	D3-6421	.5	183	1.49	1	774	2gr.	Yes	Amb.	.373
8-30	D3-6422	.5	224	1.48	1	635	2gr.	Yes	Amb.	.373
8-30	D3-6423	.5	185	1.68	1	711	2gr.	Yes	Amb.	.373
8-30	D3-6424	.5	225	1.73	1	660	2gr.	Yes	Amb.	.373
8-30	D3-6425	.5	183	1.62	1	771	2gr.	Yes	Amb.	.373
8-30	D3-6426	Biprop.	Valve	did not	open.	88	2gr.	Yes	Amb.	.373
8-30	D3-6427	Biprop.	Valve	did not	open.	63	2gr.	Yes	Amb.	.373
8-30	D3-6428	.5	201	1.66	1	643	2gr.	Yes	Amb.	.374
8-30	D3-6429	.5	205	1.67	1	714	2gr.	Yes	Amb.	.374
8-30	D3-6430	.5	206	1.67	1	746	2gr.	Yes	Amb.	.374
8-30	D3-6431	.5	206	1.66	1	541	2gr.	Yes	Amb.	.374
8-30	D3-6432	.5	205	1.64	-	54*	2gr.	Yes	Amb.	.0045
8-30	D3-6433	.5	207	1.65	1	716*	2gr.	Yes	Amb.	.374
8-30	D3-6434	.25	-	-	-	-	2gr.	Yes	Amb.	.009
8-30	D3-6435	.25	P_c=35@	Det.	1	177	2gr.	Yes	Amb.	.0115
8-31	D3-6436	.25	P_c=66@	Det.	-	531	2gr.	Yes	Amb.	.015
8-31	D3-6437	Biprop.	Valve	did not	open.	-	2gr.	Yes	Amb.	-

*Before P_c Rise

HIGH ALTITUDE IGNITION TESTS

INJECTOR S/N RDV-DD-PR-1

ALTITUDE - 250,000 FEET

	Test Date	Test No.	L.O.R.	P_c	r	Spike Duration ms	Spike Magnitude (psia)	Helium Sat.	$T_{Props.}$ °F	Recovery Time
	8-31	D3-6438	.050	200	1.6	1.1	183	Yes	40	<20 ms
	9-1	D3-6439		200	1.6	1.7	300	Yes	40	<20 ms
	9-1	D3-6440		200	1.6	1.0	261	Yes	40	<20 ms
	9-5	D3-6441		180	1.6	1.3	249	Yes	40	<20 ms
	9-5	D3-6442		180	1.6	1.3	123	Yes	40	<20 ms
	9-6	D3-6443		180	1.6	1.2	148	Yes	40	<20 ms
	9-6	D3-6444		180	1.6	.5	56	Yes	40	<20 ms
	9-6	D3-6445		180	1.6	1.1	50	Yes	40	<20 ms
	9-6	D3-6446		220	1.6	.8	171	Yes	40	<20 ms
	9-6	D3-6447		220	1.6	1.0	213	Yes	40	<20 ms
	9-6	D3-6448		220	1.6	1.1	343	Yes	40	<20 ms
	9-7	D3-6449		220	1.6	1.3	364	Yes	40	<20 ms
	9-7	D3-6450		220	1.6	1.3	192	Yes	40	<20 ms
	9-7	D3-6451		200	1.6	.8	112	Yes	40	<20 ms
	9-7	D3-6452		200	1.6	2.3	372	Yes	40	<20 ms
	9-7	D3-6453		200	1.6	.9	112	No	40	<20 ms
	9-7	D3-6454		200	1.6	2.5	387	No	40	<20 ms
	9-7	D3-6455		200	1.6	1.0	90	No	40	<20 ms
	9-8	D3-6456		200	1.6	1.5	398	No	40	<20 ms
	9-8	D3-6457	.050	200	1.6	1.1	362	No	40	<20 ms
SEA LEVEL CHECHOUT SERIES	9-11	D3-6458	5.2	175	1.56	-	-	Yes	40	
	9-11	D3-6459	5.2	176	1.56	-	-	Yes	40	
	9-11	D3-6460	5.0	178	1.71	-	-	Yes	40	Found eroded Ox. Orifices

PART B

SUMMARY OF TEST DATA

Injector S/N DD-PR-2

Barrier % (ρ) = 11.8%

INJECTOR S/N DD-PR-2 TEST RESULTS

TEST DATE	TEST NO.	L.O.R.* (SEC)	P_C (PSIA)	r (O/F)	C* (FPS)
8/1/72	D3-6382	2.0	207	1.617	5330
	D3-6383	4.9	206	1.607	5290
	D3-6384	2.0	170	1.753	5255
	D3-6385	2.2	178	1.848	5200
	D3-6386	2.0	193	1.950	5140
	D3-6387	1.8	196	1.743	5270
	D3-6388	4.9	194	1.773	5225
	D3-6389	1.8	201	1.366	5280
	D3-6390	5.0	200	1.388	5270
	D3-6391	1.9	176	1.692	5290
	D3-6392	5.0	177	1.686	5260
	D3-6393	2.0	175	1.861	5195
	D3-6394	4.9	173	1.859	5170
	D3-6395	2.0	183	1.401	5310
	D3-6396	4.9	181	1.374	5280
8/2/72	D3-6397	45.2	193	1.573	5385
	D3-6398	45.3	194	1.600	5385

*LENGTH OF RUN.

INJECTOR S/N DD-PR-2 TEST RESULTS

TEST DATE	TEST NO.	L.O.R. (SEC)	P_C (PSIA)	r (O/F)	C* (FPS)	T_{MAX}*- ^oF 9 O'CLOCK SIDE	T_{MAX}*- ^oF 3 O'CLOCK SIDE
8/14/72	B2-1380	2.3	182	1.681	5255		
	B2-1381	2.2	164	1.222	5150		
	B2-1382	2.2	186	1.665	5240		
	B2-1383	2.3	199	1.537	5270		
	B2-1384	40.2	197	1.540	5280	2078	2148
8/17/72	B2-1385	45.2	199	1.547	5290	2098	2179
	B2-1386	2.3	199	1.342	5180		
8/18/72	B2-1387	45.2	204	1.665	5260	2151	2157
9/15/72	B2-1405	2.0	212	1.645	5235		
9/18/72	B2-1406	2.0	200	1.444	5235		
	B2-1407	2.0	198	1.407	5190		
	B2-1408	2.1	199	1.507	5220		

*PYROSCANNER DATA

INJECTOR S/N DD-PR-2 TEST RESULTS

PULSE I_{SP} - $\epsilon = 31$

TEST DATE	TEST NO.	R (O/F)	I_{SP} (SEC)	EPW (MS)	FREQUENCY (CPS)
9-19-72	B2-1409	CHECK OF PULSE METERING FLOW SYSTEM			
9-19-72	B2-1410				
9-19-72	B2-1411	1.41	208.6	50	5
9-19-72	B2-1412	1.41	207.5	50	5
9-19-72	B2-1413	RAN OUT OF FUEL DURING TEST			
9-21-72	B2-1414	CHECKOUT			
9-21-72	B2-1415	1.54	169.7	50	1
9-21-72	B2-1416	1.54	168.3	50	1
9-21-72	B2-1417	1.66	222.9	50	5
9-21-72	B2-1418	1.68	220.4	50	5
9-22-72	B2-1419	1.62	200.2	90	1
9-22-72	B2-1420	1.63	199.4	90	1
9-22-72	B2-1421	1.67	234.4	170	1
9-22-72	B2-1422	1.70	229.0	170	1

INJECTOR S/N DD-PR-2 TEST RESULTS

PERFORMANCE

$\epsilon = 31$

TEST DATE	TEST NO.	LOR (SEC)	P_C (PSIA)	R (O/F)	C* (FPS)	F- (LB)	C_{F-}	I_{SP-} (SEC)
9-25-72	B2-1423	5.0	198.0	1.650	5058	596.8	1.781	279.8
9-25-72	B2-1424	10.0	198.2	1.645	5081	597.5	1.781	281.0
9-28-72	B2-1425	5.1	202.9	1.622	5110	606.2	1.767	280.5
9-28-72	B2-1426	5.1	223.4	1.611	5116	669.9	1.767	280.8
9-28-72	B2-1427	5.1	184.8	1.531	5167	552.2	1.761	282.7
9-28-72	B2-1428	5.1	182.5	1.614	5138	545.1	1.760	280.9
9-28-72	B2-1429	5.1	205.0	1.462	5169	611.3	1.758	282.1
9-29-72	B2-1430	5.1	204.5	1.509	5135	612.9	1.766	281.7
9-29-72	B2-1431	5.2	203.9	1.599	5152	611.0	1.766	282.5
9-29-72	B2-1432	5.3	202.7	1.678	5103	606.3	1.763	279.3
9-29-72	B2-1433	10.3	201.6	1.601	5093	606.8	1.774	280.7
		10.3	201.3	1.538	5087	607.2	1.778	280.8
		10.4	201.6	1.599	5096	606.9	1.974	280.9
		10.2	201.6	1.600	5097	606.3	1.773	280.5
		PULSES		50 MS - 5CPS				
		10.3	201.7	1.599	5097	606.8	1.773	280.6
		PULSES		50 MS - 2CPS				
		10.2	201.3	1.602	5086	606.2	1.775	280.3
		PULSES		50 MS - 1CPS				
		10.3	201.[illegible]	1.598	5084	605.7	1.774	280.1
		PULSES		50 MS - 5CPS				

INJECTOR S/N DD-PR-2 TEST RESULTS

PERFORMANCE/THERMAL

TEST DATE	TEST NO.	LOR (SEC)	P_C (PSIA)	R (O/F)	C* (FPS)	T_{MAX} - °F 3 O'CLOCK SIDE	T_{MAX} - °F 9 O'CLOCK SIDE
10-4-72	D3-6463	1.9	198	1.686	5015	–	–
10-4-72	D3-6464	2.0	201	1.637	5100	–	–
10-4-72	D3-6465	30.1	201	1.631	5150	2106	<1850
10-4-72	D3-6466	29.7	202	1.610	5170	2165	<1850
10-4-72	D3-6467	29.8	201	1.609	5150	2143	<1550
10-6-72	D3-6468	4.9	203	1.592	5175	–	–
10-6-72	D3-6469	99.7	192	1.589	5160	WORST CASE MDC	
		99.7	192	1.579	5165		
		99.6	192	1.573	5180		
		99.7	192	1.566	5175		
		99.8	192	1.564	5170		
		99.8	191	1.553	5175		
		99.8	194	1.608	5175		
		99.8	193	1.582	5155		
		99.6	192	1.568	5171		
		99.5	192	1.554	5165		
10-6-72	D3-6470	600.8	199	1.615	5175	ENDURANCE TEST	
			199	1.615	5170		
			199	1.614	5170		
			198	1.606	5170		
			198	1.597	5180		
			198	1.603	5175		
			198	1.606	5170		
			198	1.611	5770		
			198	1.610	5175		

INJECTOR S/N DD-PR-2 TEST RESULTS

WORST CASE MDC-HELIUM SATURATED PROPELLANTS

TEST DATE	TEST NO.	LOR (SEC)	P_C (PSIA)	R (O/F)	C* (FPS)
10-9-72	D3-6471	100.5	10 PULSES + 100 SEC.		
	D3-6472	100.3	198	1.598	5135
		100.4	198	1.585	5155
		100.4	199	1.585	5166
		100.3	199	1.582	5165
		100.3	199	1.572	5154
		100.1	200	1.573	5163
		100.2	201	1.579	5171
		100.5	202	1.571	5176
		100.5	203	1.571	5181
		100.4	203	1.569	5180

INJECTOR S/N DD-PR-2 TEST RESULTS

e = 31

TEST DATE	TEST NO.	LOR (SEC)	R (O/F)	Pc (PSIA)	C* (FPS)	F∞ (LB)	$C_{F\infty}$	$I_{SP\infty}$ (SEC)
9-29-72	B2-1434	50		PULSE HEATING		1000 PULSES - 50 MS - 5 CPS		
	B2-1435	10				200 PULSES - 50 MS - 1 CPS		
	B2-1436	18				200 PULSES - 90 MS - 1 CPS		
	B2-1437	34				200 PULSES -170 MS - 1 CPS		
10-13-72	B2-1438	10.3	1.568	205.1	5177	612.2	1.765	283.7
	B2-1439	5.4	1.452	207.2	5155	619.7	1.769	283.1
	B2-1440	5.4	1.644	205.1	5156	615.3	1.773	284.0
	B2-1441	5.1	1.473	190.0	5187	567.9	1.768	284.7
	B2-1442	5.1	1.532	188.2	5190	563.4	1.770	285.3
	B2-1443	5.1	1.540	232.3	5177	696.3	1.772	284.8
	B2-1444	30.5	1.615	199.1	5194	592.6	1.760	283.9
10-16-72	B2-1445	4.7		PULSE I_{SP}		94 PULSES - 50 MS - 1 CPS		
	B2-1446	4.6				92 PULSES - 50 MS - 1 CPS		
	B2-1447	4.9				98 PULSES - 50 MS - 1 CPS		
	B2-1448	5.0				100 PULSES - 50 MS - 1 CPS		
	B2-1449	4.4				89 PULSES - 50 MS - 1 CPS		
	B2-1450	4.4				89 PULSES - 50 MS - 1 CPS		
10-18-72	B2-1451*	5.0				100 PULSES - 50 MS - 1 CPS		
	B2-1452*	5.0						
	B2-1453*	5.0						
	B2-1454*	5.0						
	B2-1455*	5.0						
	B2-1456*	5.0						
10-23-72	B2-1457▽	2.2	1.637	218	5115	646.8	1.767	280.8
	B2-1458▽	2.3	1.541	197	5090	583.0	1.762	278.5
	B2-1459▽	2.3	1.590	204	5090	606.0	1.764	279.0

* With helium saturated propellants.
▽ With Moog Valve.

CONTINUED NEXT PAGE.

INJECTOR S/N DD-PR-2 TEST RESULTS

$\epsilon = 31$

TEST DATE	TEST NO.	LOR (SEC)	R (O/F)	P_c (PSIA)	C* (FPS)	F_∞ (LB)	C_{F_∞}	I_{SP_∞} (SEC)
10-24-72	B2-1460 ▽	5.0		PULSE ISP		100 PULSES - 50 MS - 1 CPS		
	B2-1461 ▽	4.8		↓		96 PULSES - 50 MS - 1 CPS		
	B2-146[illegible] ▽	3.5		↓		100 PULSES - 35 MS - 1 CPS		
	B2-1463 ▽	3.5		↓		100 PULSES - 35 MS - 1 CPS		
	B2-1464 ▽	2.[illegible]		↓		89 PULSES - 27 MS - 1 CPS		
	B2-1465 ▽	1.[illegible]		↓		68 PULSES - 27 MS - 1 CPS		
	B2-1466 ▽	1.9		↓		71 PULSES - 27 MS - 1 CPS		
	B2-1467 ▽	2.1		↓		76 PULSES - 27 MS - 1 CPS		
	B2-1468 ▽	4.4		↓		88 PULSES - 50 MS - 1 CPS		
	B2-1469 ▽	.5		↓		10 PULSES - 50 MS - 1 CPS		
	B2-1470 ▽	.5		↓		10 PULSES - 50 MS - 1 CPS		
10-27-72	B2-1471*	5.2	1.609	203	5087	607.1	1.772	279.[illegible]
	B2-1472*			PULSE ISP		100 PULSES - 50 MS - 5 CPS		
10-30-72	B2-1473*			↓		100 PULSES - 50 MS - 5 CPS		
	B2-1474*			↓		100 PULSES - 50 MS - 1 CPS		
	B2-1475*			↓		100 PULSES - 50 MS - 1 CPS		
	B2-1476*			↓		50 PULSES - 90 MS - 1 CPS		
	B2-1477*	4.5		↓		50 PULSES - 90 MS - 1 CPS		
10-31-72	B2-1478*	4.25		↓		25 PULSES -170 MS - 1 CPS		
	B2-1479*	4.25		↓		25 PULSES -170 MS - 1 CPS		
	B2-1480*	5.0		↓		100 PULSES - 50 MS - 5 CPS		
11-3-72	B2-1481 ◇	5.0		↓		100 PULSES - 50 MS - 1 CPS		
	B2-1482 ◇	5.0		↓		100 PULSES - 50 MS - 1 CPS		
	B2-1483 ◇	5.0		↓		100 PULSES - 50 MS - 1 CPS		
	B2-1484 ◇	5.0		↓		100 PULSES - 50 MS - [illegible] CPS		
	B2-1485 ◇	5.0		↓		100 PULSES - 50 MS - [illegible] CPS		
	B2-1486 ◇	5.0		↓		100 PULSES - 50 MS - [illegible] CPS		
11-7-72	B2-1487* ◇	5.0		↓		100 PULSES - 50 MS - [illegible] CPS		
	B2-1488* ◇	5.0		↓		100 PULSES - 50 MS - [illegible] CPS		
11-8-72	B2-1489* ◇	5.0		↓		100 PULSES - 50 MS - [illegible] CPS		
	B2-1490* ◇	5.0		↓		100 PULSES - 50 MS - [illegible] CPS		
	B2-1491* ◇	5.0		↓		100 PULSES - 50 MS - [illegible] CPS		

◇ Cold Propellants ~ approx. 40°F

* With Helium saturated propellants

▽ With Moog Valve

CONTINUED NEXT PAGE

INJECTOR S/N DD-PR-2 TEST RESULTS

e = 31

TEST DATE	TEST NO.	LOR (SEC)	R (O/F)	P_c (PSIA)	C* (FPS)	F_∞ (LB)	$C_{F\infty}$	$I_{SP\infty}$ (SEC)
11-9-72	B2-1492 □	5.0	PULSE I_{SP}			100	PULSES - 50 MS - 1 CPS	
	B2-1493 □	5.0				100	PULSES - 50 MS - 1 CPS	
	B2-1494 □	5.0				100	PULSES - 50 MS - 1 CPS	
	B2-1495 □	5.0				100	PULSES - 50 MS - 1 CPS	
	B2-1496 □	4.5				90	PULSES - 50 MS - 1 CPS	
11-10-72	B2-1497 □	4.8				95	PULSES - 50 MS - 1 CPS	
11-13-72	B2-1498 □	5.0				100	PULSES - 50 MS - 1 CPS	
	B2-1499 □	4.2				84	PULSES - 50 MS - 1 CPS	
	B2-1500 □	4.0				80	PULSES - 50 MS - 1 CPS	
	B2-1501	4.9	1.712	205	5225	619.8	1.784	289.4
	B2-1502	5.1	1.703	205	5215	617.8	1.781	288.4
	B2-1503	5.2	1.56	199	5190	600.2	1.779	286.8
	B2-1504	5.3	1.56	200	5170	600.7	1.779	285.6
11-14-72	B2-1505	5.2	1.627	202	5175	607.2	1.781	286.2
	B2-1506	5.3	1.594	199	5120	604.3	1.796	285.7
11-16-72	B2-1507* □	5.0	PULSE I_{SP}			100	PULSES - 50 MS - 1 CPS	
11-17-72	B2-1508* □	5.0				100	PULSES - 50 MS - 1 CPS	
	B2-1509* □	4.8				95	PULSES - 50 MS - 1 CPS	
11-20-72	B2-1510* □	5.0				99	PULSES - 50 MS - 1 CPS	
11-21-72	B2-1511	5.0				100	PULSES - 50 MS - 1 CPS	
	B2-1512	5.0				100	PULSES - 50 MS - 1 CPS	

□ Hot Propellants ~ approx. 120°F.

* With Helium saturated propellants.

Date 1972	Test # B-2	Hel. Sat. Prop	O/F	P_c PSIA	Prop. Temp. °F	EPW MS	Freq. CPS	$I_{T\infty}$ Lb-Sec.	$I_{sp\infty}$ Sec.	Remarks
10-27	1472	Yes	1.68	200	Amb	50	5	26.5	200.4	
10-30	1473	Yes	1.68	200	Amb	50	5	26.6	203.5	
10-30	1474	Yes	1.58	200	Amb	50	1	24.3	168.9	
10-30	1475	Yes	1.62	200	Amb	50	1	23.9	167.2	
10-30	1476	Yes	1.66	200	Amb	90	1	49.2	204.2	
10-30	1477	Yes	1.69	200	Amb	90	1	49.1	208.5	
10-30	1478	Yes	1.72	200	Amb	170	1	97.9	233.2	
10-30	1479	Yes	1.73	200	Amb	170	1	98.9	236.6	
10-31	1480	Yes	1.80	200	Amb	50	5	27.3	197.9	
11-3	1481	No	1.62	200	40	50	1	22.4	154.9	
11-3	1482	No	1.62	200	40	50	1	22.7	157.7	
11-6	1483	No	1.62	180	40	50	1	19.6	148.8	
11-6	1484	No	1.55	180	40	50	1	19.3	150.7	
11-6	1485	No	1.69	220	40	50	1	25.4	168.7	
11-6	1486	No	1.69	220	40	50	1	24.7	167.2	
11-7	1487	Yes	1.6	200	40	50	1	-	-	
11-7	1488	Yes	1.6	200	40	50	1	-	-	
11-8	1489	Yes	1.68	200	40	50	1	22.8	164.8	
11-8	1490	Yes	1.65	180	40	50	1	19.7	155.2	
11-8	1491	Yes	1.75	220	40	50	1	26.2	168.9	
11-9	1492	No	1.54	200	110	50	1	24.2	178.1	
11-9	1493	No	1.59	200	110	50	1	23.8	171.6	
11-9	1494	No	1.36	180	110	50	1	21.1	171.0	
11-9	1495	No	1.45	180	110	50	1	21.2	160.0	
11-9	1496	No	1.31	220	110	50	1	26.7	173.5	

Bell Aerospace Company

PART C

SUMMARY OF TEST DATA

INJECTOR S/N RDV-DD-PR-2

INJECTOR S/N RDV-DD-PR-2 STEADY STATE DATA

Thrust Chamber S/N P-2E ϵ = 31 (Where thrust reported)

			TEST DATA							Pyroscanner Data (^oF)	
Date (1972)	Test	Duration (Sec.)	O/F	P_c (psia)	C* (Ft/Sec)	C* (%)	F_∞ (LBF)	$I_{sp\infty}$ (Sec)	FT (^oF)	3:00 Throat	9:00 Throat
12/1	B-2 1513	5.0	1.835	227	5289	92.4	681.5	293.7	77		
	1514	5.1	1.565	198.5	5321	93.1	589.4	292.6	78		
	1515	5.1	1.661	200.3	5333	93.1	595.6	293.7	78		
	1516	5.2	1.344	199.1	5337	94.5	587.7	291.8	78		
	1517	5.2	1.517	181.2	5330	93.4	534.5	291.2	78		
	1518	5.[illegible]	1.503	222.6	5326	93.3	658.9	292.0	78		
12/12	D-3 6479	5.0	1.692	194.1	5329	93.0			60		
12/13	6480	21.4	1.592	198.6	5324	93.0			57		
									58		
									58	2241	2508
12/14	6481	30.3	1.446	197.4	5280	92.8			56		
									56		
									58	2608	2317
	6482	30.2	1.623	198.1	5288	92.4			55		
									56		
									56	2335	2449
	6483	30.3	1.475	198.5	5270	92.5			54		
									54		
									56	2288	2260
	6484	30.2	1.629	179.2	5278	92.2			54		
									54		
									56	2116	2231
	6485	30.0	1.790	200.2	5300	92.5			54		
									54		
									55	2483	2506
	6486	30.1	1.628	221.8	5321	92.9			54		
									54		
									54	2422	2486

SUMMARY OF PULSE DATA

INJECTOR S/N RDV-DD-PR-2

$\epsilon = 31$

ALTITUDE TESTS

Test # B-2	Hel. Sat. Prop.	O/F	P_c psia	Prop. Temp. ^oF	EPW ms	Freq. cps	$I_{T\infty}$ Lb-Sec.	$I_{sp\infty}$ Sec.
1521	No	1.52	200	81	50	1	24.5	183.5
1522	No	1.51	200	83	50	1	24.8	-
1423	No	1.51	200	93	50	1	25.2	185.6
1524	No	1.55	200	93	90	1	50.4	224.7
1525	No	1.53	200	103	170	1	97.0	247.8
1526	No	1.54	200	106	35	1	16.7	158.9
1527	No	1.55	200	106	50	1	26.0	188.6
1528	No	1.56	200	110	90	1	50.2	222.3
1529	No	1.53	200	110	170	1	96.9	246.5
1530	No	1.55	200	108	35	1	17.2	162.6

PART D

SUMMARY OF TEST DATA

INJECTOR DD-BB

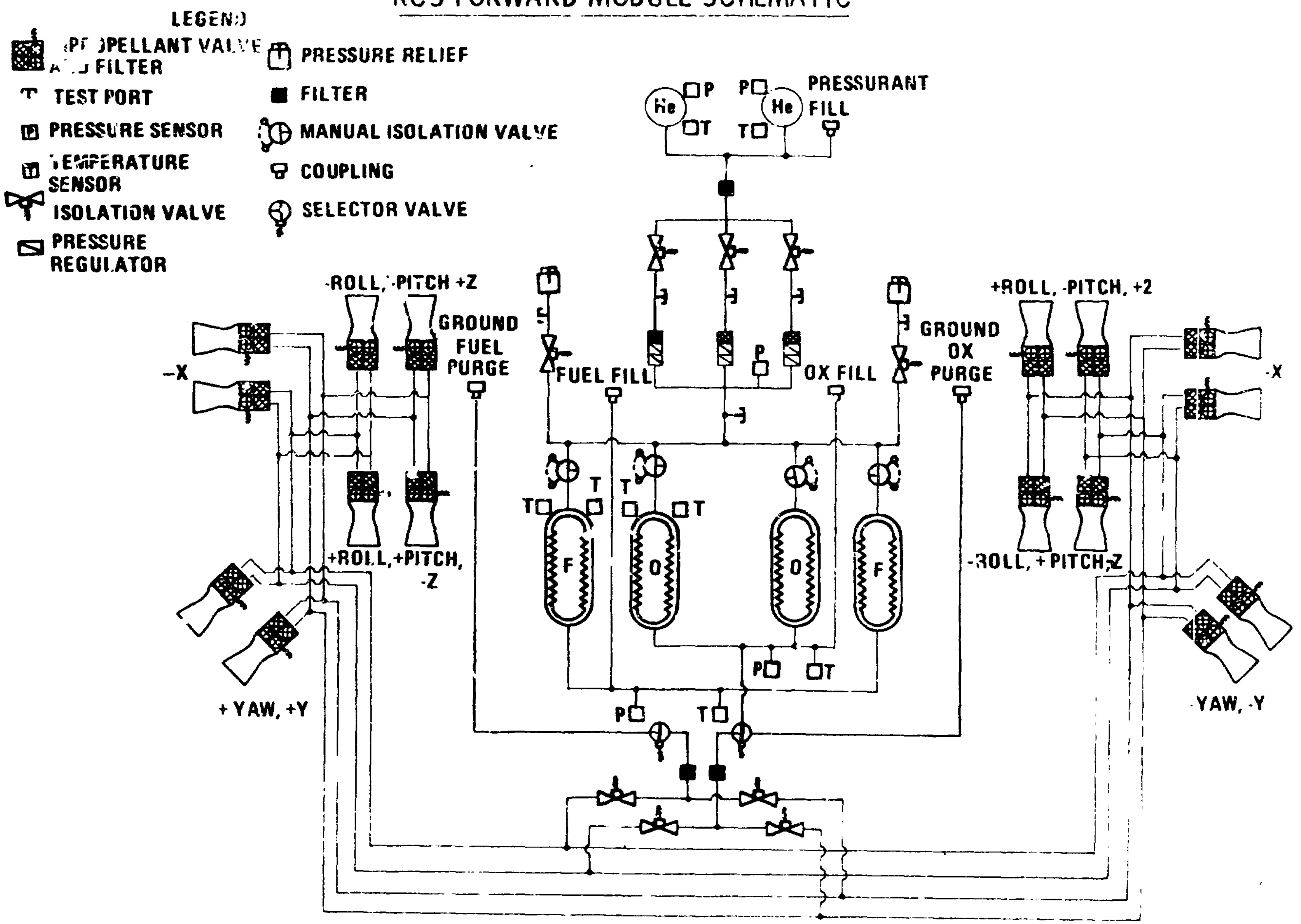
INJECTOR DD-BB TEST RESULTS

Test # D3-	P_c psia	O/F	C*	Plugged Orifices	Chamber Temp. °F
6373	197.3	1.596	5377	None	2244/2351
6399	173.3	1.686	5306	20% Adjacent barrier orifices plugged.	C/O
6400	172.3	1.671	5303	" "	C/O
6401	171.7	1.637	5319	" "	2717
6402	171.6	1.662	5330	33%	C/O
6403	172.6	1.676	5328	33%	2832/2662
6404	170.5	1.664	5137	2 Outer Adjacent Fuel Primary Orifices Plugged	C/O
6405	168.7	1.659	5143	" "	2465/2227

APPENDIX III

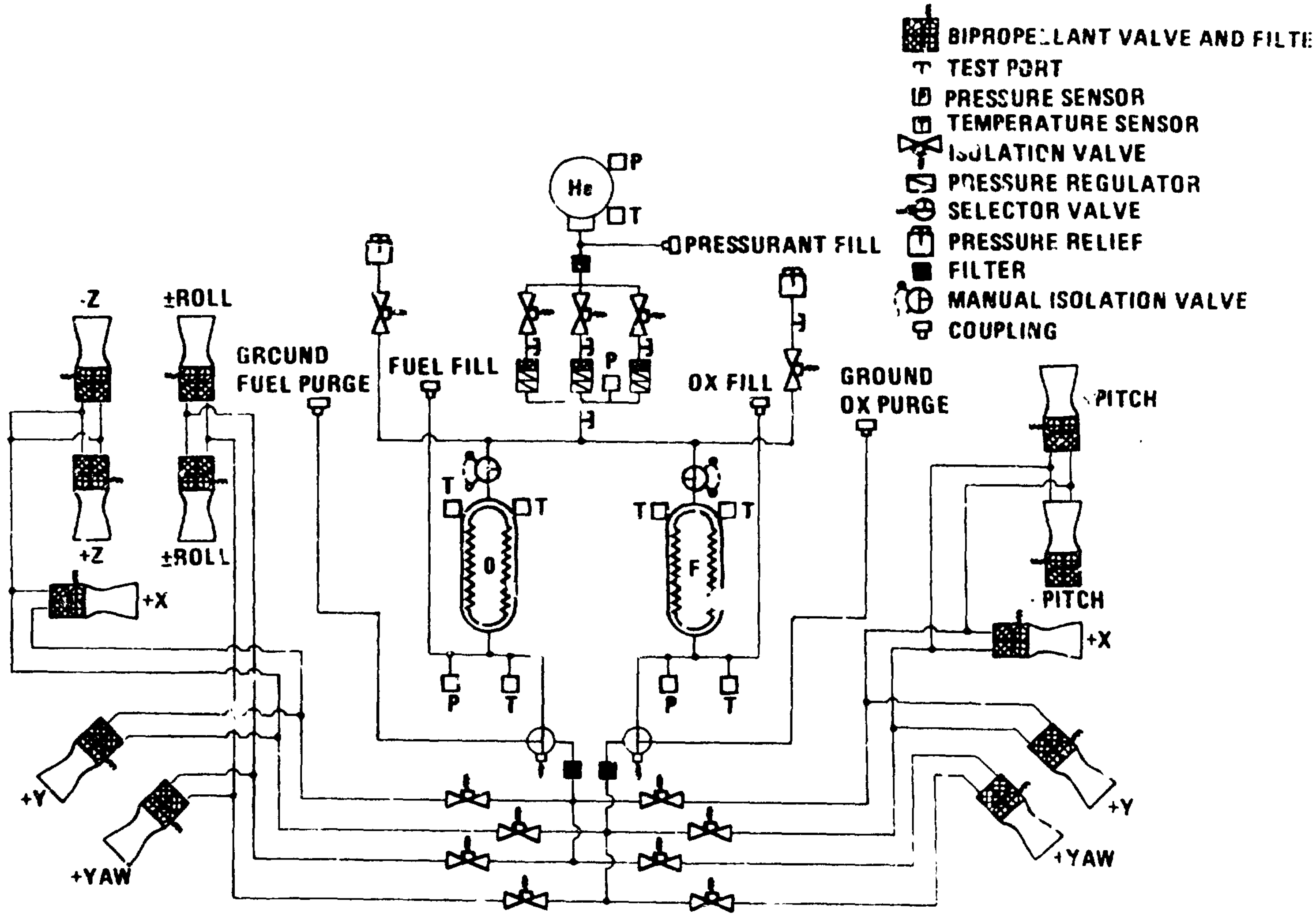
SYSTEMS ANALYSIS

RCS FORWARD MODULE SCHEMATIC*



*ASSUMED CONFIGURATION FOR ANALYSIS PURPOSES.

RCS LEFT OR RIGHT AFT MODULE SCHEMATIC *

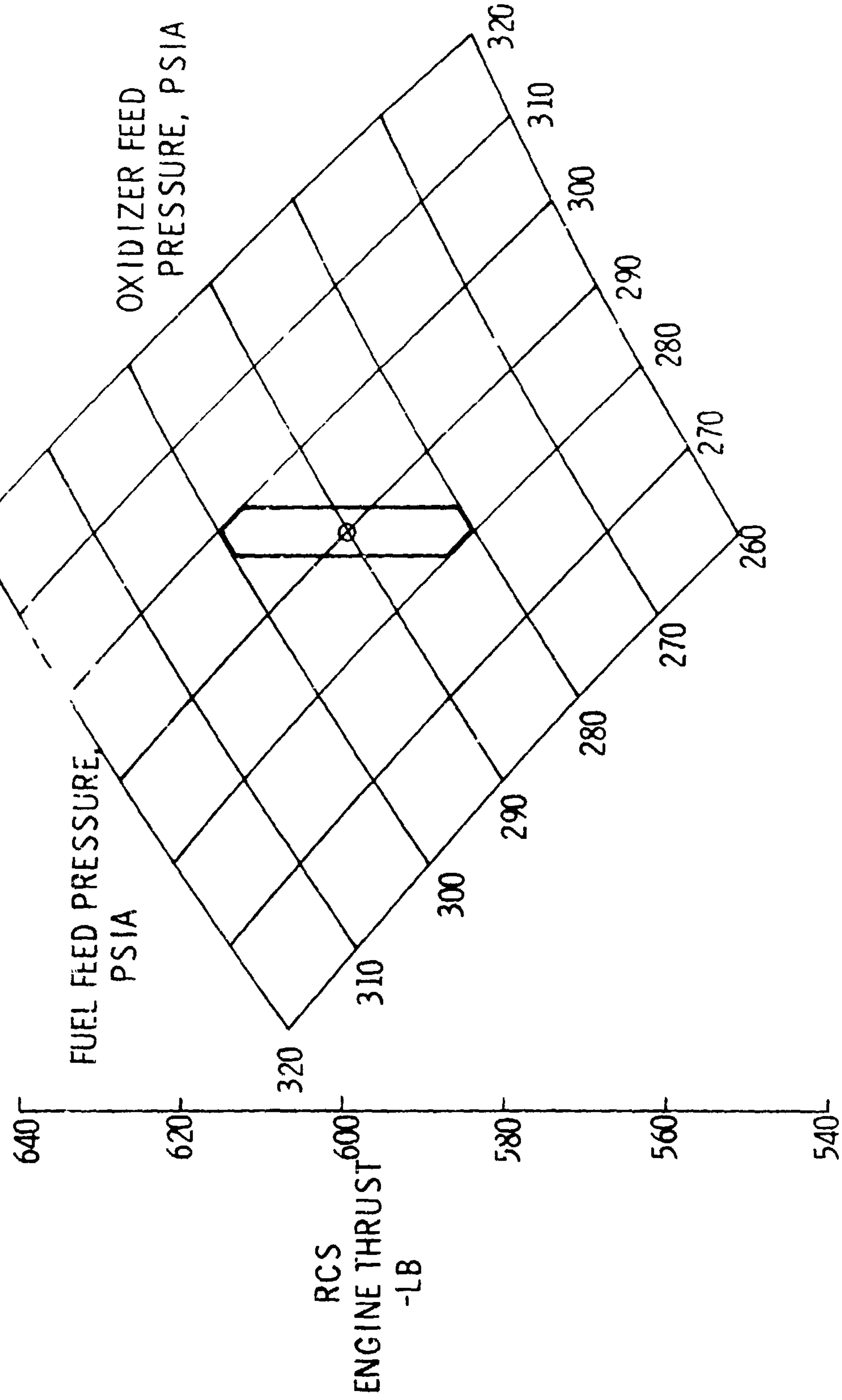


*ASSUMED CONFIGURATION FOR ANALYSIS PURPOSES.

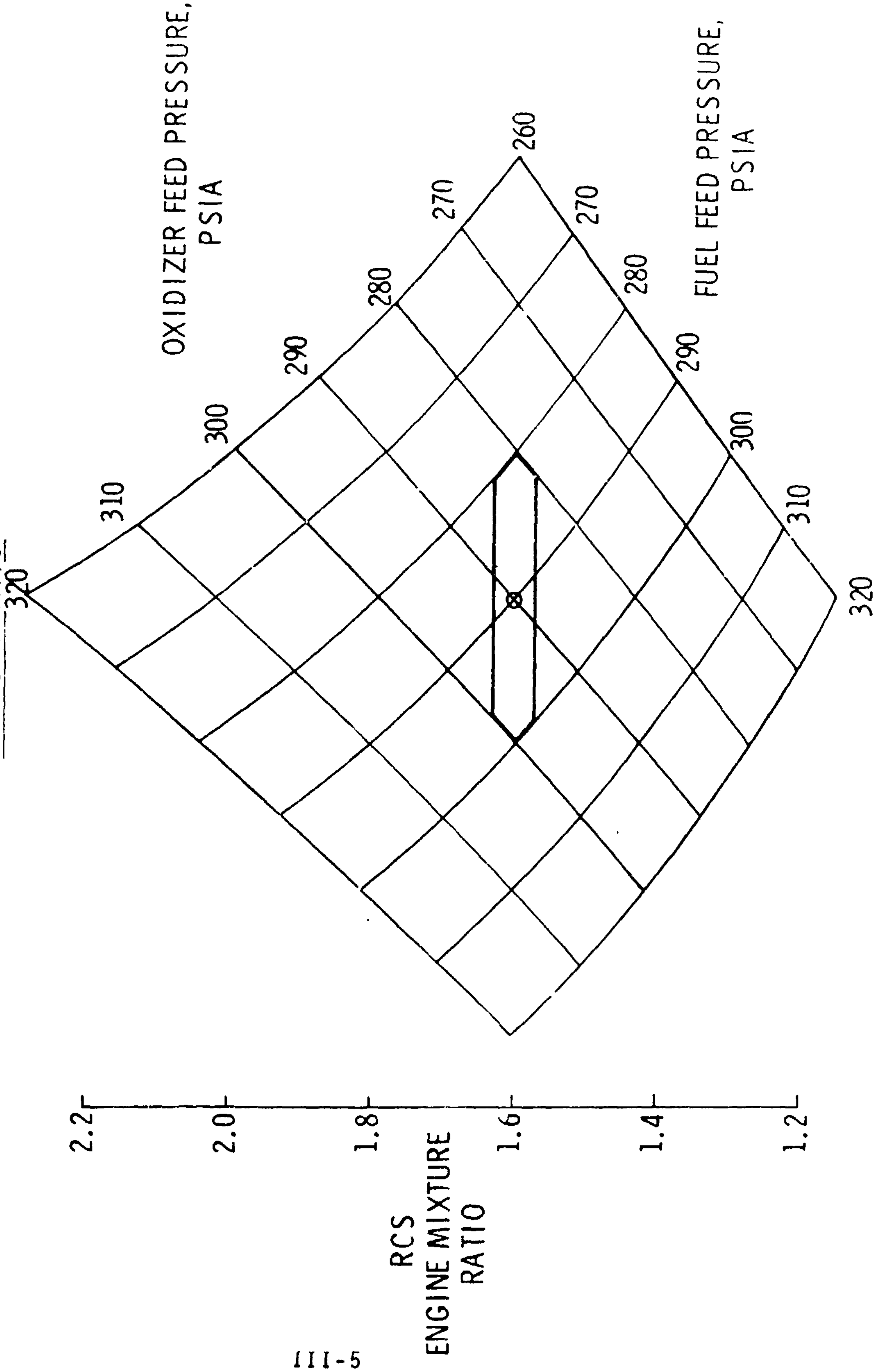
PRESSURE SCHEDULE - NOMINAL CONDITIONS - 2 ENGINES FIRING

	FUEL		OX
ENGINE CHAMBER		200.	
INJECTOR DROP	50.		50.
ORIFICE DROP	10.		10.
VALVE DROP	30.		30.
ENGINE INLET	290.		290.
FEED LINE DROP	1.5		1.5
PROPELLANT TANK	291.		291.
GAS MANIFOLD DROP	1.		1.
PRIMARY REG OUT		292.5	
SECONDARY REG OUT		300.	
REG LOCKUP PRESSURE		297.	
RELIEF VALVE RESEAT		315.	
MAX RELIEF PRESSURE		355.	
TANK WORKING PRESSURE		355.	
MIN REG INLET		500.	
HIGH PRESS LINE DROP		50.	
MINIMUM SOURCE		550.	
HELIUM STORAGE PRESSURES			
MINIMUM AT 40. DEG F		3788.	
NOMINAL AT 68 DEG F		4000.	
MAXIMUM AT 110. DEG F		4318.	
PREPRESSURE AT 110. DEG F		277.	

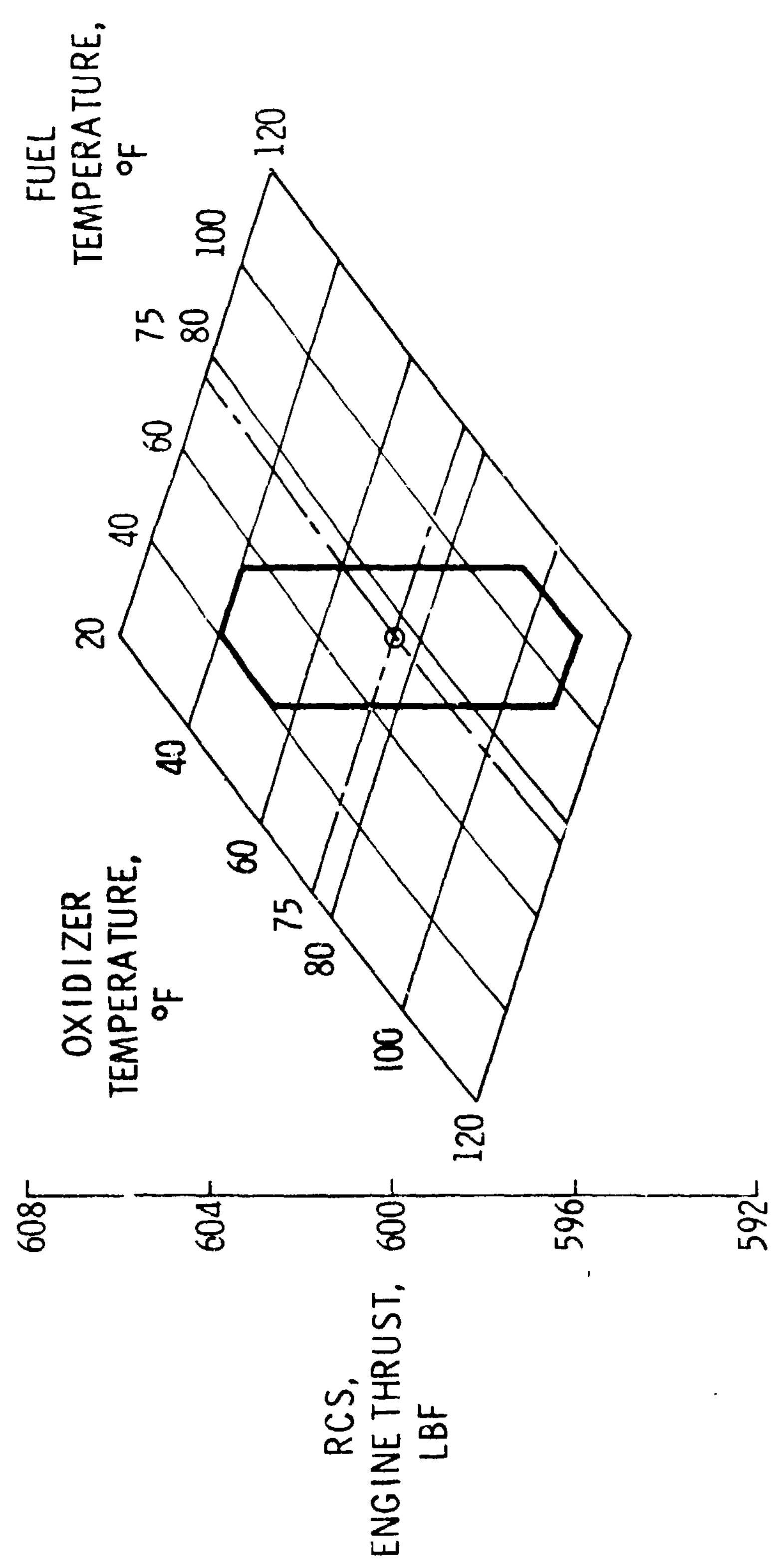
EFFECT OF FEED PRESSURE VARIATIONS ON RCS ENGINE THRUST



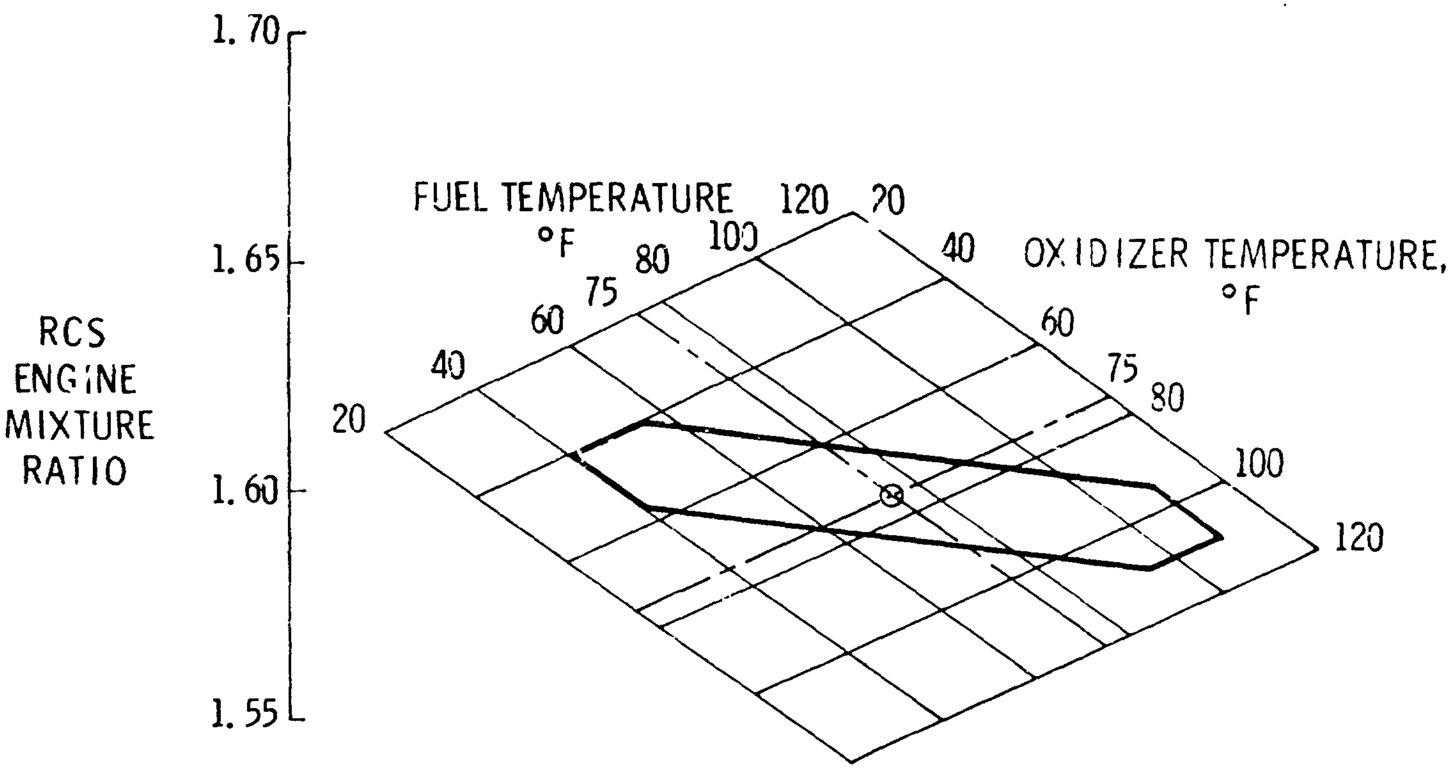
EFFECT OF FEED PRESSURE VARIATION ON RCS ENGINE MIXTURE RATIO



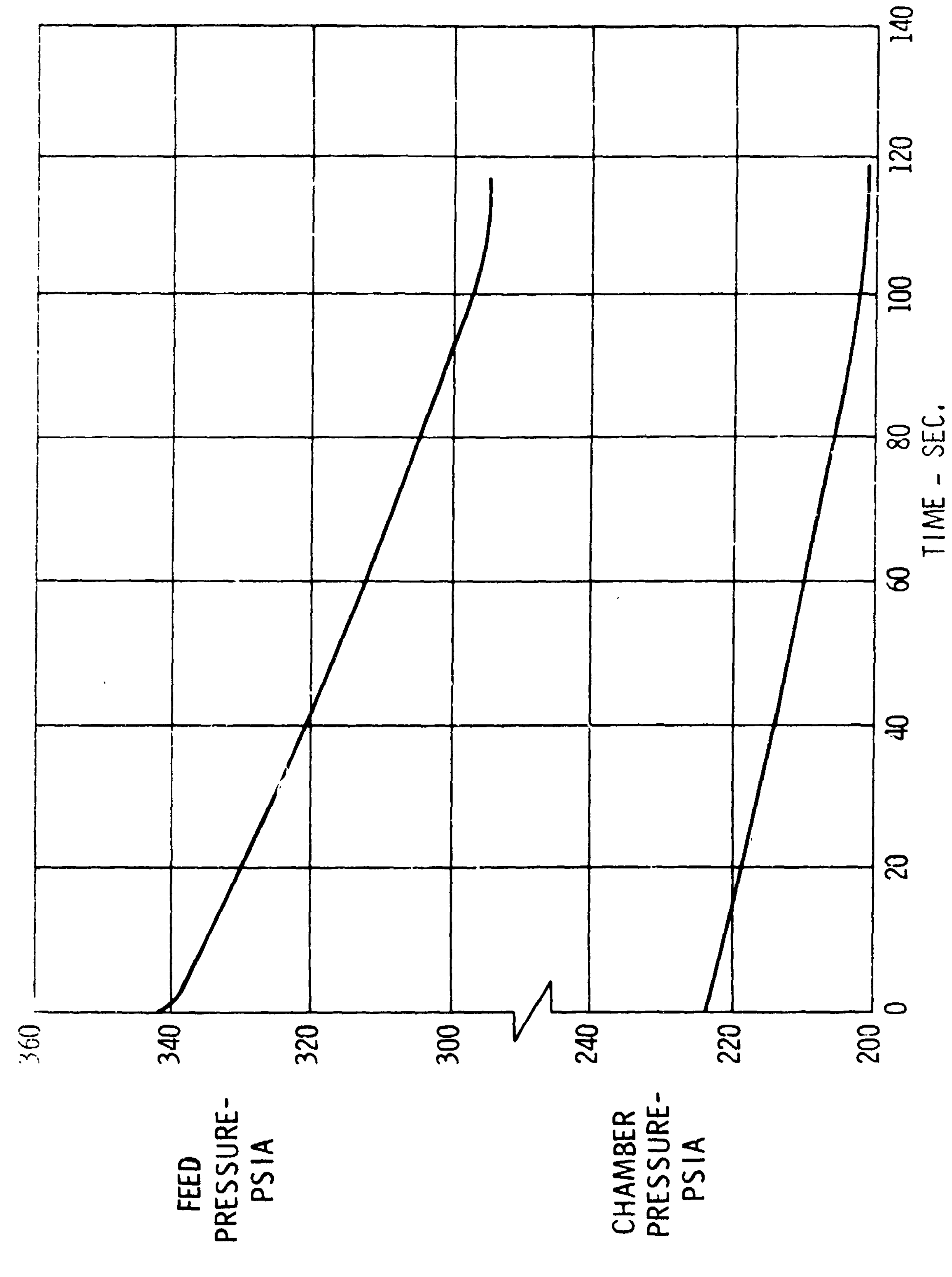
EFFECT OF PROPELLANT TEMPERATURE VARIATION ON RCS ENGINE THRUST



EFFECT OF PROPELLANT TEMPERATURE VARIATION ON RCS ENGINE MIXTURE RATIO

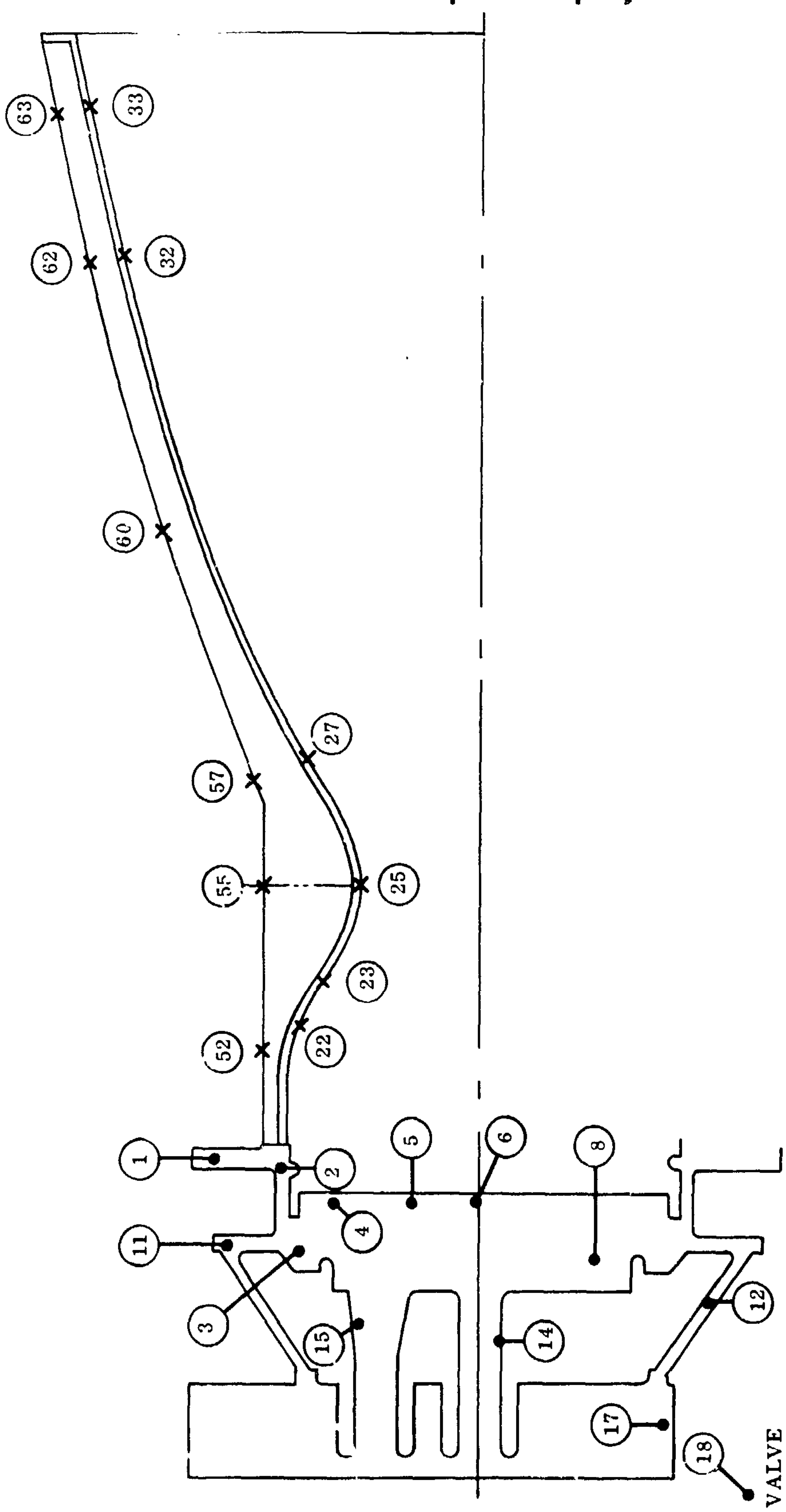


BLOW DOWN PERFORMANCE OF RCS ENGINE

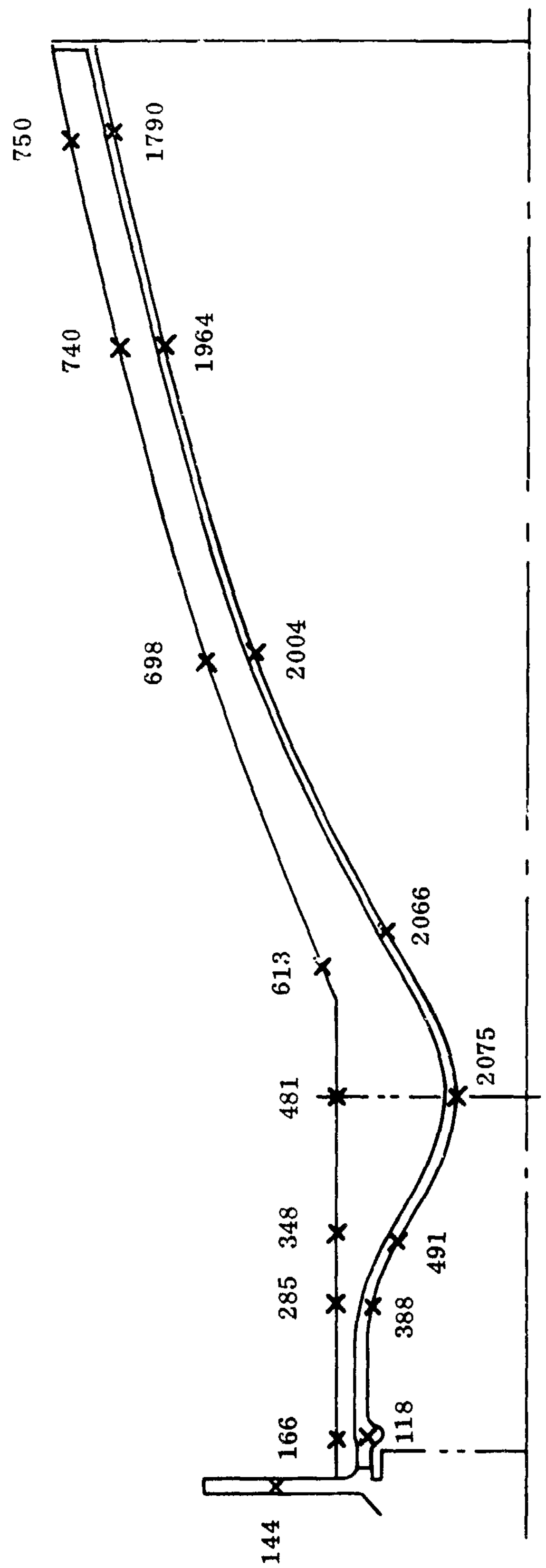


APPENDIX IV

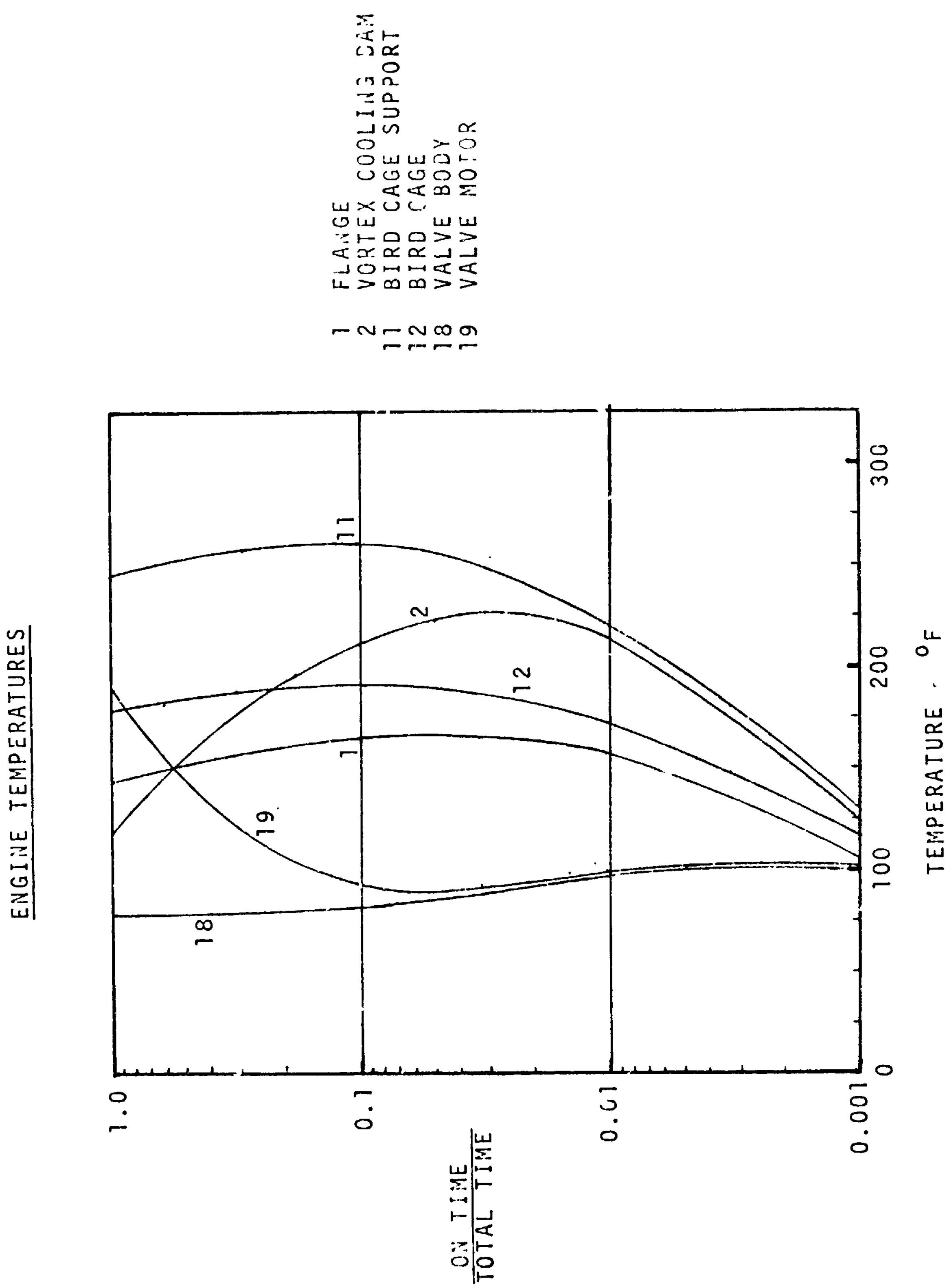
THERMAL ANALYSIS



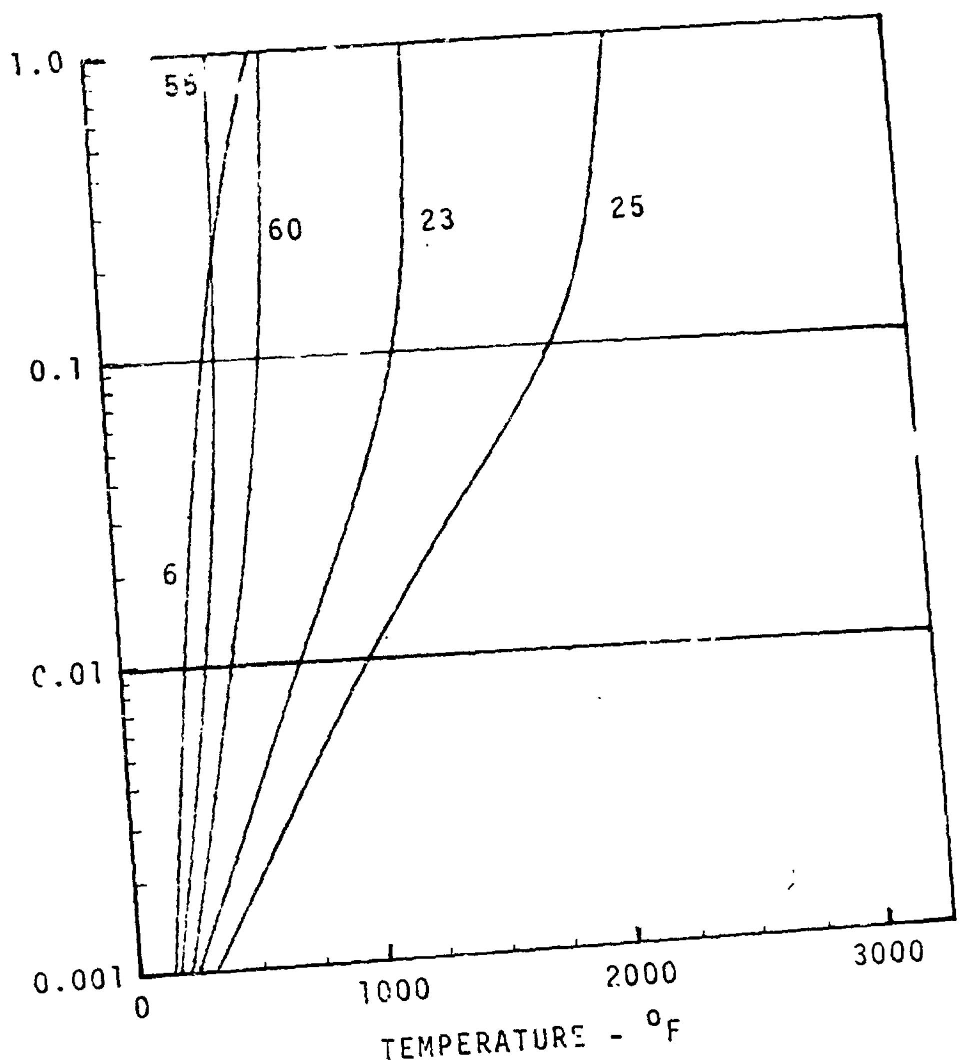
THERMAL ANALYSIS NODE LOCATIONS - FLIGHT TYPE ENGINE



STEADY STATE TEMPERATURE DISTRIBUTION - FLIGHT TYPE ENGINE

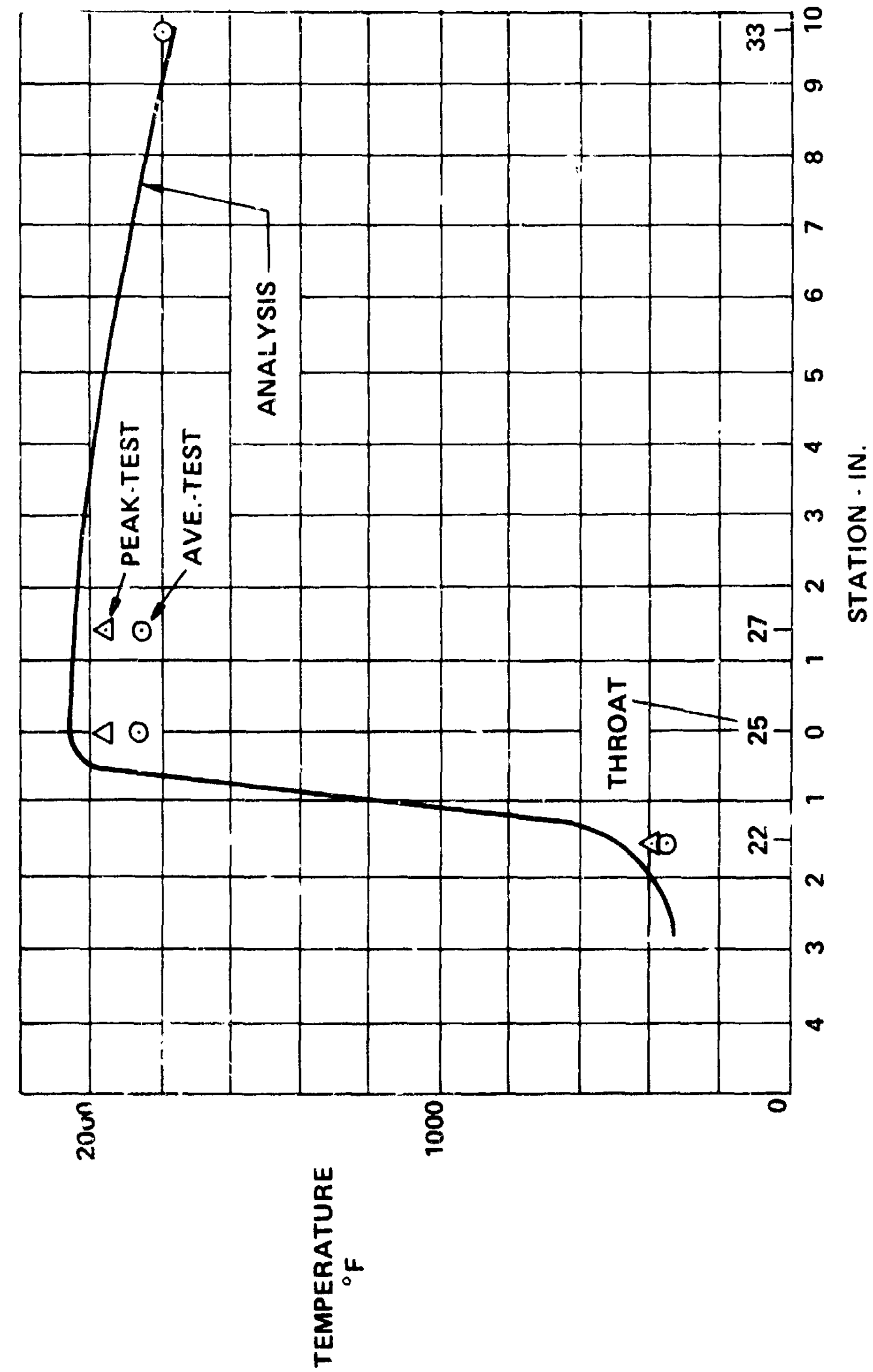

ENGINE TEMPERATURES
1 FLANGE
2 VORTEX COOLING DAM
11 BIRD CAGE SUPPORT
12 BIRD CAGE
18 VALVE BODY
19 VALVE MOTOR
11
2
12
1
19
18
1.0
0.1
0.01
0.001
0
100
200
300
TEMPERATURE, °F
ON TIME / TOTAL TIME

ENGINE WALL TEMPERATURES

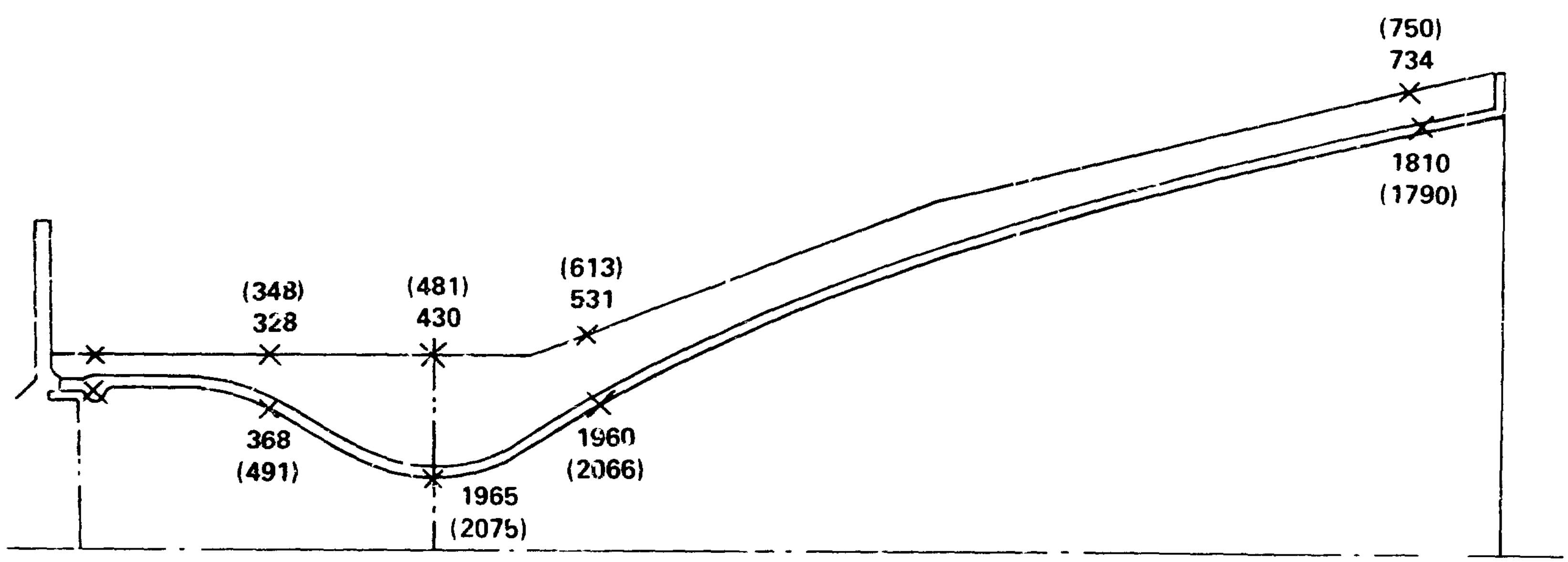


6 INJECTOR FACE
23 CONVERGING SECTION
25 THROAT WALL
55 THROAT EXTERIOR
60 NOZZLE EXTERIOR

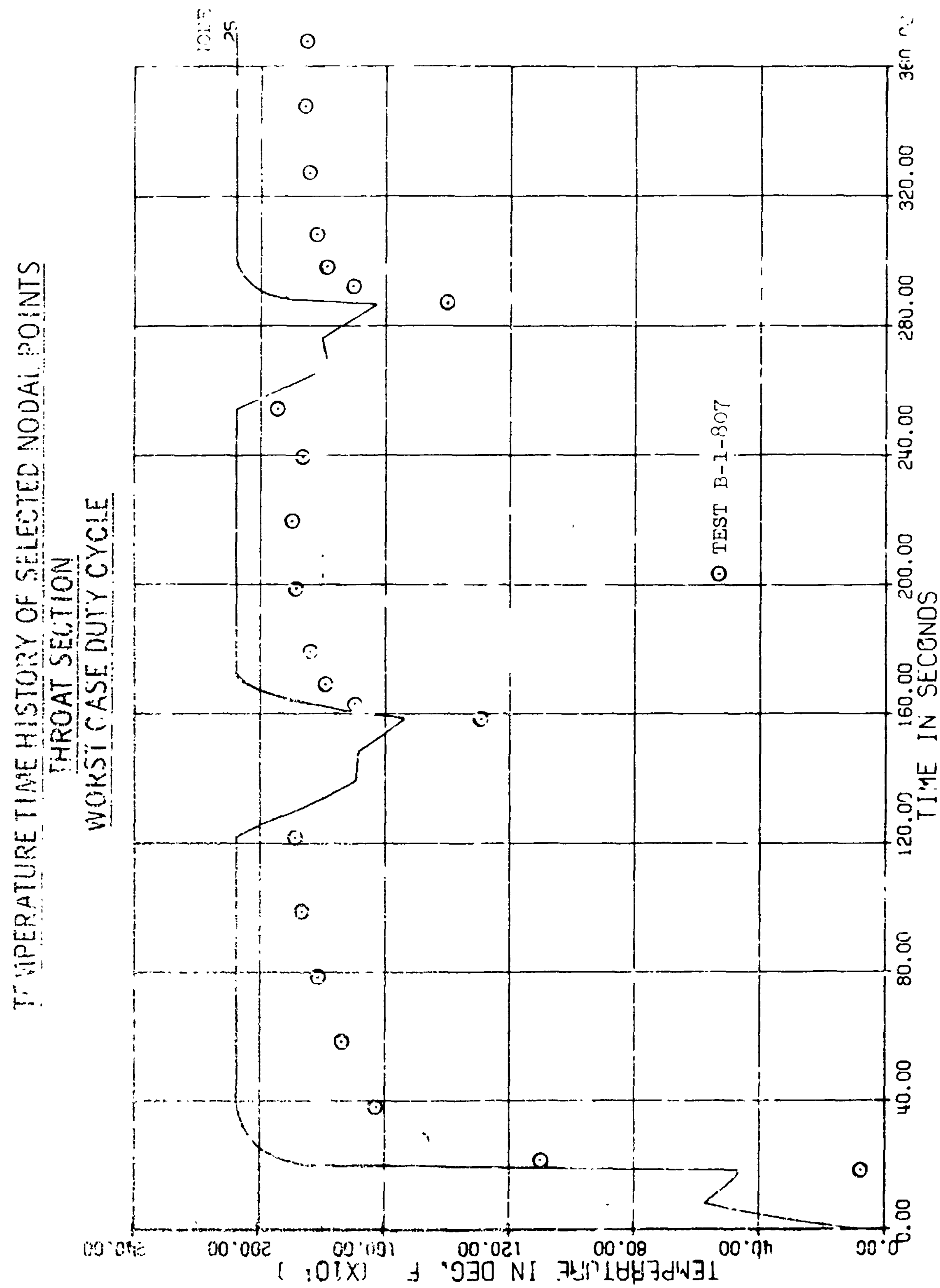
STEADY STATE THRUST CHAMBER WALL TEMPERATURE



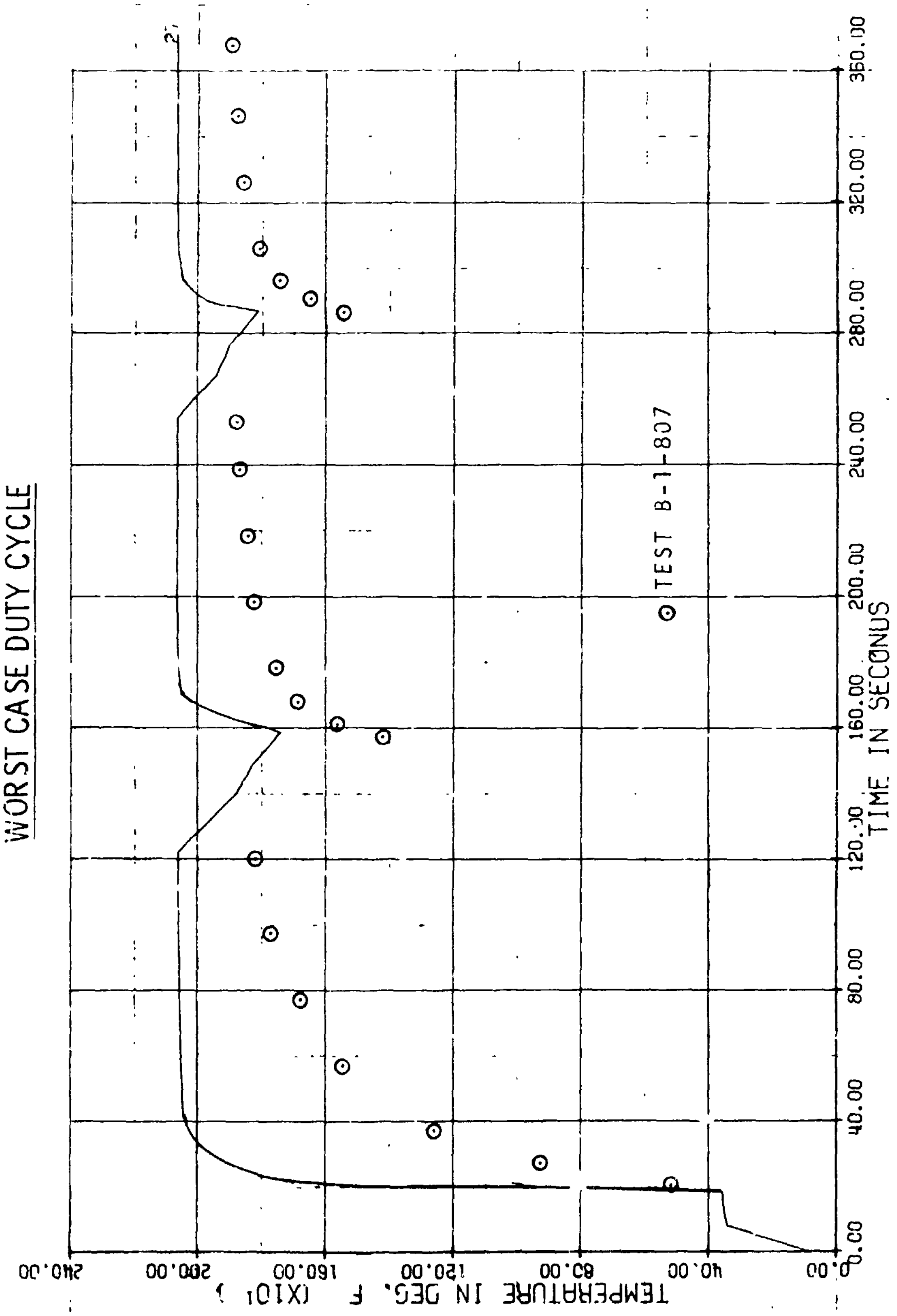
STEADY STATE TEMPERATURE DISTRIBUTION FLIGHT TYPE ENGINE - RDV-2



XXX TEST DATA - AVERAGE TEMPERATURE
(XXX) ANALYSIS


TEMPERATURE TIME HISTORY OF SELECTED NODAL POINTS
THROAT SECTION
WORST CASE DUTY CYCLE
TEST B-1-807
TIME IN SECONDS
TEMPERATURE IN DEG. F (X10¹)
0.00
40.00
80.00
120.00
160.00
200.00
240.00
280.00
320.00
360.00

TEMPERATURE TIME HISTORY OF SELECTED NODAL POINTS
DIVERGING SECTION
WORST CASE DUTY CYCLE



APPENDIX V

RELIABILITY

FAILURE MODE ANALYSES

- FAILURE MODE - GENERAL NATURE OF THE FAILURE
- CAUSE - DETAILED NATURE OF THE FAILURE
- CLASSIFICATION

  CRITICAL - FAILURE CAUSING MISSION ABORT OR SAFETY HAZARD
  MAJOR - FAILURE DEGRADING RELIABILITY OR PERFORMANCE OF THE SYSTEM
  MINOR - FAILURE HAVING NO SIGNIFICANT EFFECT ON RELIABILITY OR PERFORMANCE

- SYMPTOMS - INFORMATION DETECTABLE IN FLIGHT
- ACTION - METHODOLOGY REQUIRED TO MAXIMIZE MISSION/HARDWARE SAFETY
- EFFECTS

  ENGINE/FLIGHT HARDWARE
  MISSION
  GROUND HANDLING PERSONNEL AND PROCEDURES

- PREVENTION

  DESIGN FEATURES
  ACCEPTANCE TEST
  POST FLIGHT SERVICE/INSPECTION

PRELIMINARY FAILURE MODE AND CONTROL ANALYSIS - ENGINE

Failure Mode	Cause	Class	In Flight Symptom(s)	Mission Action	Effect: Hardware	Effect: Mission	Effect: Ground Handling	Prevention/Control: Design	Prevention/Control: Acceptance Test	Prevention/Control: Ground Service Test (Tentative)
External Leakage (Propellant)	Failure of one of the engine to valve interface static seals or of the injector manifold weld	Major	A reduction in the engine's thrust. Vapors in engine compartment	Desirable to isolate engine if detected	An off mixture ratio and reduced thrust will occur. Continued operation not likely to be detrimental.	Reduced ability to deliver specified total impulse. Possible termination of the engine's use due to improper operation. Increased contamination in vicinity of vehicle	Possible personnel hazard. Engine should be replaced	Double (redundant) seals are used in both the oxidizer and fuel manifolds Weld Design – Proven in past experience and RCS development Fully inspectable – Minimum in number	Leak checked at 1.6 times operating pressure Proof Pressure Tested Welds are inspected – Optical (at 20X minimum) – X ray (radiographic) – Ultrasonic No cracks or porosity permitted	Visual inspection may reveal problem Leakage tests should be performed leak test solution – vapor detectors
	Failure of both engine to valve interface static seals (oxidizer and fuel) or of both injector feed tubes (oxidizer and fuel) or manifold failure	Critical	Reduced engine thrust. Erratic chamber pressure vapors and heat in engine compartment	Isolate engine immediately upon detection	Local hypergolic reaction and combustion may result in engine or system components damage. Continued operation likely to aggravate situation	Reduced engine thrust. Termination of the engine use. Possible premature termination of the mission	Possible personnel hazard. Engine must be replaced	Double (redundant) seals are used in both the oxidizer and fuel manifolds Weld Design – Proven in past experience and RCS development – Fully inspectable – Minimum in numbers	Leak checked at 1.6 times operating pressure Proof Pressure Tested Welds are inspected – Optical (at 20X minimum) – X ray (radiographic) – Ultrasonic No cracks or porosity permitted	Visual inspection should reveal problem Leak tests should be performed
External Leakage at Injector to Chamber Joint (combustion products)	Failure of injector to chamber joint	Critical	Reduced engine thrust. Erratic chamber pressure vapors and heat in engine compartment	Isolate engine immediately upon detection	– Hot combustion gases escaping beyond engine shell may cause damage to system components – Continued operation highly detrimental	– Reduced engine thrust – Termination of the engine's use – Possible premature termination of the mission	Possible personnel hazard. Engine must be replaced	Weld Design – Proven in past experience and RCS development – Fully inspectable – In very cool area	– The injector to chamber weld is gas pressure tested at two times operating pressure – Proof pressure tested – Weld inspected – optical (at 20X minimum) X ray Ultrasonic (No cracks or porosity permitted)	Visual inspection should reveal problem Leak test not likely to be feasible
Orifice Plugging	Internal contamination in the engine manifold or injector	Major	Reduced or erratic thrust. Excessive skin temperature	Desirable to isolate or disable engine if detected	May cause off mixture ratio operation with attendant thrust reduction	Reduced ability to deliver specified impulse. Possible termination of the engine's use.	No personnel hazard if biprop valve unimpaired	Filters are located at the inlets to the bipropellant valve to prevent downstream contamination	Cleanliness controls will be used during engine and system buildup Contamination check will be made	Evidence of orifice plugging should be detected during visual inspection
					May cause partial or complete removal of outer fuel barrier for cooling which may result in excessive skin temperatures or combustion chamber burnout	Combustion zone wall burn out would result in termination of engine's use	– Engine must be replaced. – System should be checked for contamination source	Initial test shows that contamination from nozzle end not to be a problem Further tests are planned		Evidence of orifice plugging should be detected during visual inspection
Excessive Skin Temperature	Improper operation of fuel film barrier	Major	High engine skin temperature	Desirable to isolate or disable engine if detected	May cause off mixture ratio operation with attendant thrust reduction.	Reduced ability to deliver specified impulse. Possible termination of the engine's use.	No personnel hazard if biprop valve unimpaired	Film cooling concept has demonstrated effectivity and reliability	Proper cooling can be demonstrated during hot fire acceptance test	
					May cause partial or complete removal of outer fuel barrier for cooling which may result in excessive skin temperatures or combustion chamber burnout	Combustion zone wall burn out would result in termination of engine's use	– Engine must be replaced – System should be checked for contamination source			
	Degradation of external insulation		High engine skin temperature	Desirable to isolate or disable engine if detected	Could result in damage or degradation to surrounding structure or equipment No damage to engine	Depends on system effect	No hazard – Engine should be refurbished or replaced	Insulation design will be proven during development	– Visual inspection [illegible] proper fabrication and installation – Operation verified during fire test	Visual inspection should reveal problems

PRELIMINARY FAILURE MODE AND CONTROL ANALYSIS - ENGINE (CONTD)

Failure Mode	Cause	Class	In Flight Symptom(s)	Mission Action	Effect: Hardware	Effect: Mission	Effect: Ground Handling	Prevention/Control: Design	Prevention/Control: Acceptance Test	Ground Service Test (Tentative)
Combustion Instability (High Frequency)	Change in engine operating conditions or ignition delay	Major	Erratic thrust Unusual vibration High skin temperature	Shutdown engine immediately upon detection Isolate and/or disable valve if incident is repeated	May result in structural failure of the chamber or supporting structure	Inability to deliver specified total impulse in proper orientation Possible termination of engine's use Possible termination of mission	No personnel hazard if biprop valve unimpaired Incident would have global implications necessary repair/replacement action(s) cannot be anticipated	Engine design has demonstrated insensitivity to instability when tested with detonation devices and at conditions intended to produce instability Further stability tests planned during development	Bomb damping could be verified during hot fire acceptance tests	Visual inspection to assure damper in good condition
Structural Failure of Thrust Chamber	Overstress of thrust chamber's metal	Major	Cracking or destruction of thrust chamber	Shutdown and disable valve and/or isolate engine immediately upon detection	Destruction of thrust chamber Damage to surrounding structure and equipment	Termination of engine's use Possible termination of mission	No personnel hazard if biprop valve unimpaired Engine must be replaced	Thrust chamber design is based on adequate safety margins for pressure and structural stress	Proof pressure test Hot fire test	Incipient cracks may be detected by visual inspection
Structural Failure of Engine Mount	Overstress of engine mount	Critical	Change in engine thrust vector Unusual vibration Vapors and heat in engine compartment	Shutdown and disable valve and or isolate engine immediately upon detection	Propellant inlet lines may be broken Possible damage to surrounding structure and equipment	Inability to deliver specified total impulse in proper orientation Termination of engine's use Possible termination of mission	No personnel hazard if biprop valve unimpaired Engine must be replaced	The engine mount's design is based on adequate structural safety margin	Nondestructive tests to assure integrity of mount	Incipient cracks or mis alignment may be detected by visual inspection
Hot Gas Leakage at Instrumentation Port	Insufficient or failed seal	Major	Heat vapors in engine compartment Excessive local skin temperature Erratic instrument reading	Desirable to disable engine	Dependent upon amount of hot gas leakage, possible damage to - Instrument - Engine - Surrounding Structure	Possible termination of engine's use	No personnel hazard unless other areas, e.g., biprop valve, plumbing, damaged Engine should be repaired or replaced	Materials selection and mechanical design will minimize possibilities of leakage Any welds involved will use proven designs and will be fully inspectable	The engine is leak checked at 1.6 times operating pressure using N_2 gas. No leakage is allowed at the thrust chamber pressure pick off or pressure transducer Welds will be inspected - Visually - X ray - Ultrasonic	Visual inspection should reveal minor leakage before it becomes problem Leak test not likely to be feasible

PRELIMINARY FAILURE MODE AND CONTROL ANALYSIS - VALVE

Failure Mode	Cause	Class	In Flight Symptom(s)	Mission Action	Effect			Prevention Control		
					Hardware	Mission	Ground Handling	Design	Acceptance Test	Ground Service Test (Tentative)
Internal Leakage past the Valve Seat (Propellant)	Contamination	Major	Possible improper engine ignition would occur when commanded Some ignition or venting when engine is off Unusual vibration	If detected engine may be isolated	Possible damage to injector due to improper ignition	Increased contamination in vicinity of vehicle Loss of RCS propellant(s) Extraneous impulse applied to vehicle Possible termination of use of engine	Possible safety hazard to personnel Engine should be replaced	Filter elements (20/3[illegible] micron) are located at the valve inlet ports. Rigid contamination controls are maintained throughout fabrication assembly and valve testing The valve seat is made of Teflon and the flapper button is made of metal. This arrangement tends to compensate for minor contamination at the valve seat. Small contamination particles tend to become imbedded in the Teflon with flapper actuation thus reducing the leakage path opening between flapper and seat	Internal leakage tests using gas (N_2) are performed as part of the acceptance tests. These tests require that the leakage (N_2 gas) at either valve outlet port shall not exceed 10 cc/hr when monitored for 12 minutes with 300 psig applied at both inlet ports Additional leakage tests are performed at the thrust chamber acceptance test level	Leakage tests should be performed using vapor detectors
External Leakage from Valve (Propellant)	Failure of one of the seals at the interface between valve body and outlet manifold	Major	Reduced engine thrust Vapors in engine compartment	Desirable to isolate engine if detected	Continued operation not likely to be detrimental	An off mixture ratio and reduced thrust would occur Possible termination of the engine's use Increased contamination in vicinity of vehicle	Possible personnel hazard Engine should be replaced	The valve design uses welded joints to the maximum extent possible The only mechanical seals used in the valve are seals at the body and outlet manifold interface Redundant seals are employed	An external leakage test is performed as part of the valve proof pressure test. No visible indication of leakage is allowed when N_2 gas at 450 psig is applied simultaneously to the valve inlet and outlet ports The proof pressure test is performed at 1.5 times the design operating pressure Additional leakage tests are performed at the thrust chamber acceptance test level	Leakage tests should be performed using vapor detectors
	Failure of both seals at the interface between valve body and outlet manifold (oxidizer and fuel)	Major	Reduced engine thrust Vapors and heat in the engine compartment	Isolate engine immediately on detection.	Leakage of both oxidizer and fuel would result in hypergolic reaction The valve may become damaged Continued operation likely to aggravate situation	Reduced thrust. Termination of engines use Possible mission termination. Increased contamination in vicinity of vehicle.	Possible personnel hazard Engine must be replaced			
External Leakage from Valve (Propellant)	Fracture of one Flexure Sleeve	Major	Reduced engine thrust Vapors in engine compartment Abnormal valve electrical behavior	Desirable to isolate engine if detected	Leakage of oxidizer or fuel externally from the valve would result in a reduction of the engine's thrust and also loss of propellant overboard	An off mixture ratio and reduced thrust would occur Possible termination of the engine's use	A reduction in engine thrust would be noted during ground firing if leakage were sufficient Leakage stains would be noted during ground inspection Engine would be replaced	The valve armature, flexure tubes, and the flappers are a completely welded assembly The flexure tube concept has been proven to be reliable from the results obtained from the manufacturer of bipropellant valves for the Minuteman program	An external leakage test is performed as part of the valve proof pressure test. No visible indication of leakage is allowed when N_2 gas at 450 psig is applied simultaneously to the valve inlet and outlet ports The proof pressure test is performed at 1.5 times the design operating pressure Additional leakage tests are performed at the thrust chamber acceptance test level	Leakage tests should be performed using vapor detectors
	Fracture of both Flexure Sleeves	Major	Possible explosion Reduced engine thrust Vapors and heat in engine compartment Possible extraneous thrust	Isolate engine immediately upon detection	Continued operation likely to aggravate situation Leakage of both oxidizer and fuel would result in hypergolic reaction The valve may become damaged	Termination of engine's use Possible mission termination Increased contamination in vicinity of vehicle	Possible personnel hazard Engine must be replaced			

PRELIMINARY FAILURE MODE AND CONTROL ANALYSIS - VALVE (CONTD)

Failure Mode	Cause	Class	In Flight Symptom(s)	Mission Action	Effect: Hardware	Effect: Mission	Effect: Ground Handling	Prevention Control: Design	Prevention Control: Acceptance Test	Prevention Control: Ground Service Test (Tentative)
Failure of Valve to Open	Contamination or Valve Torque Motor Failure	Major	Engine will not fire when commanded Abnormal valve electrical behavior	Isolate and/or disable engine	No further hardware damage is likely	Loss of propellant overboard when only one side opens. Engine's use would be terminated Increased contamination in vicinity of vehicle (if only one side opens)	Engine must be replaced No personnel hazard	Rigid contamination controls are maintained throughout fabrication, assembly and testing of the valve Filter elements (20/35 micron) are located at the valve inlet ports to prevent contamination entry through the inlet ports and into the shutoff points The valve design contains no sliding parts The materials wetted by the propellants are 17 7 PH corrosion resistant steel and 304L corrosion resistant steel These materials were chosen because of their excellent compatibility with the propellant The permanent magnets in the torque motor are Alinco This material was selected to benefit from the high energy product The torque motor coils use a high temperature ML and incapsulated with a resilient compound This construction permits thermal expansions and provides a cushion for the coils during exposure to shock and vibration	Tests to determine that the valve actuates properly are included in the valve acceptance test The thrust chamber acceptance test includes valve cycling	Additional periodic valve actuation tests will be performed as part of ground servicing
Failure of Valve to Close	Contamination, Flexure Sleeve Failure	Major	Impulse and chamber pressure when not commanded Abnormal valve electrical behavior	Isolate engine	Further hardware damage not likely	Increased contamination in vicinity of vehicle Loss of propellant overboard Engine's use would be terminated	Engine must be replaced Possible personnel hazard.	Rigid contamination controls are maintained throughout fabrication assembly and testing of the valve Filter elements (20/35 micron) are located at the valve inlet ports to prevent contamination entry through the inlet ports and into the shutoff points. The valve design contains no sliding parts The materials wetted by the propellants are 17 7 PH corrosion resistant steel and 304L corrosion resistant steel These materials were chosen because of their excellent compatibility with the propellant The permanent magnets in the torque motor are Alinco This material was selected to benefit from the high energy product The torque motor coils use a high temperature ML and incapsulated with a resilient compound This construction permits thermal expansions and provides a cushion for the coils during exposure to shock and vibration And, the valve is normally closed due to the magnetic bias of the torque motor so that interruption of electrical power will leave the valve closed	Tests to determine that the valve actuates properly are included in the valve acceptance test The thrust chamber acceptance test includes valve cycling	Additional periodic valve actuation tests will be performed as part of ground servicing

PRELIMINARY FAILURE MODE AND CONTROL ANALYSIS - VALVE (CONTD)

Failure Mode	Cause	Class	In Flight Symptom(s)	Mission Action	Effect: Hardware	Effect: Mission	Effect: Ground Handling	Prevention Control: Design	Prevention Control: Acceptance Test	Ground Service Test (Tentative)
Failure of Valve to Close Fully	Contamination	Major	Impulse and chamber pressure when not commanded Abnormal valve electrical behavior	Isolate engine	Further hardware damage not likely	Increased contamination in vicinity of vehicle Loss of propellant overboard Engine's use would be terminated. Engine may be usable dependent upon thrust level and start and stop response of engine ignition	Engine must be replaced Possible personnel hazard	Rigid contamination controls are maintained throughout fabrication, assembly and testing of the valve Filter elements (20/35 micron) located at the valve inlet ports to prevent contamination entry through the inlet ports and into the shutoff points. The valve design contains no sliding parts The materials wetted by the propellants are 17-7 PH corrosion resistant steel and 304L corrosion resistant steel. These materials were chosen because of their excellent compatibility with the propellant The permanent magnets in the torque motor are Alnico. This material was selected to benefit from the high energy product. The torque motor coils use a high temperature MI and incapsulated with a resilient compound. This construction permits thermal expansions and provides a cushion for the coils during exposure to shock and vibration And, the valve is normally closed due to the magnetic bias of the torque motor so that interruption of electrical power will leave the valve closed	Tests to determine that the valve actuates properly are included in the valve acceptance test The thrust chamber acceptance test includes valve cycling	Additional periodic valve actuation test, will be performed as part of ground servicing
Change in Propellant Pressure Drop Through Valve	Fuel and/or Oxidizer Flow Path Altered due to Contamination or Change in Valve Seat Opening Dimensions	Major	Reduced engine thrust Abnormal valve electrical behavior	None or disable if severe loss of thrust occurs	If oxidizer and/or fuel flows are offset sufficiently, engine erosion or burn out will result. Otherwise, no further damage is likely	Possible termination of engine use	Engine should be replaced No personnel hazard	Rigid contamination controls are maintained throughout fabrication assembly and testing of the valve The valve flapper stops are individually adjusted during valve assembly to achieve the proper orifice opening required. The flapper stops, used for this adjustment, are welded in place	The valve acceptance test includes functional testing, requiring not only cycling the valve, but also specific flow rates. The thrust chamber assembly acceptance test also requires valve cycling and specific pressure drops	None
Delayed Valve Opening	Contamination or Bent Flexure Sleeve	Major	Impaired impulse performance Abnormal valve electrical behavior	None, unless guidance requires	No effect except valve will open slowly	Delay in attaining rated thrust level The navigation equipment may have to adjust the engine firing duration to allow for slow thrust buildup	Possible replacement of engine assembly No personnel hazard	Filter elements (20/35 micron) are located at the valve inlet ports upstream from the opening/shutoff ports Rigid contamination controls are maintained throughout fabrication assembly and valve testing The valve design contains no sliding parts and all clearances are relatively large minimizing its susceptibility to contamination	Valve response tests as part of the valve acceptance test require the valve to open fully in 0.030 second or less Valve response tests are also included in the thrust chamber assembly acceptance test	Valve response tests
Delayed Valve Closing	Contamination or Bent Flexure Sleeve	Major	Larger than normal shutdown impulse Abnormal valve electrical behavior	None, unless guidance requires	No effect except valve will close slowly	Delayed thrust chamber shutdown causing excessive residual impulse Compensation must be made to the engine firing duration, by the navigation equipment to adjust for residual impulse	Possible replacement of engine assembly No personnel hazard	Filter elements (20/35 micron) are located at the valve inlet ports upstream from the opening/shutoff ports Rigid contamination controls are maintained throughout fabrication assembly and valve testing The valve design contains no sliding parts and all clearances are relatively large minimizing its susceptibility to contamination	Valve response tests as part of the valve acceptance test require the valve to close fully in 0.20 second or less Valve response tests are also included in the thrust chamber assembly acceptance test	Valve response tests

PRELIMINARY FAILURE MODE AND CONTROL ANALYSIS - VALVE (CONT'D)

Failure Mode	Cause	Class	In Flight Symptom(s)	Mission Action	Effect			Prevention Control		
					Hardware	Mission	Ground Handling	Design	Acceptance Test	Ground Service Test (Tentative)
Valve Restricted Flow (Filter Clogging)	Valve Inlet Filter Clogged	Major	Reduced engine thrust or engine fails to fire Normal valve electrical behavior	If severe enough, disable engine	An off mixture ratio and reduced thrust will occur No damage likely	Reduced ability of engine to deliver specified total impulse Possible termination of the engine's use due to improper operation	Engine should be replaced	System design/service problem	The valve acceptance test includes a flow and pressure drop test which requires the valve to have specific flow rates (oxidizer and fuel) and pressure drops Thrust chamber assembly acceptance tests require the complete assembly to have specific performance requirements for flow rate and mixture ratio	None

APPENDIX VI

PHASE III

TEST DATA

PART A

INJECTOR TEST DATA

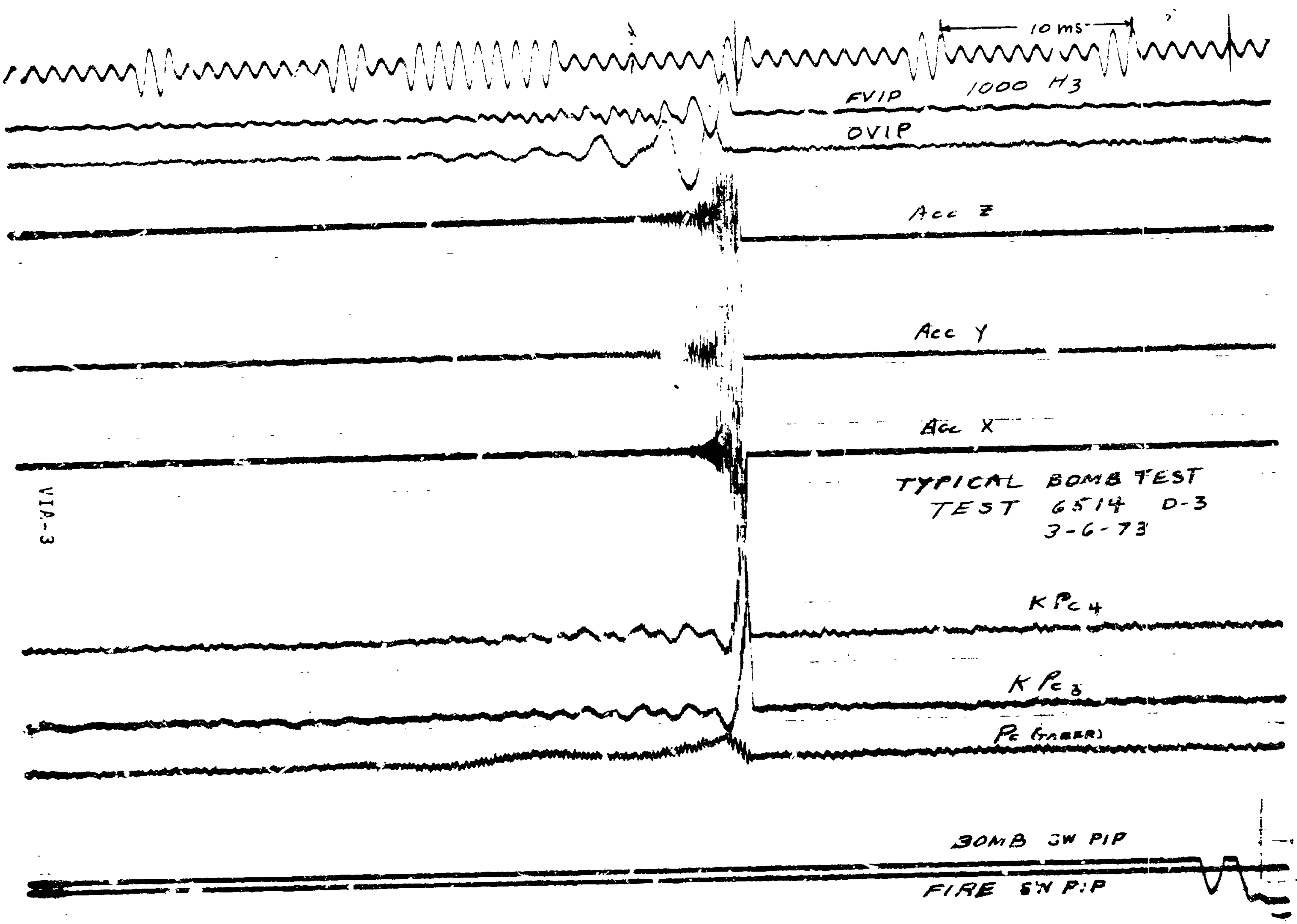
BOMB STABILITY TEST SUMMARY

INJECTOR S/N FT-1

Test No.	O/F	P_c - psia	Peak Pressure psia	Damp Time ms
D-3-6512	1.66	208	583	1.0
D-3-6513	1.59	208	612	1.0
D-3-6514	1.71	202	721	1.0
D-3-6515	1.61	218	746	1.0
D-3-6516	1.63	173	567	0.9
D-3-6517	1.64	204	585	1.1

INJECTOR S/N FT-2

Test No.	O/F	P_c - psia	Peak Pressure psia	Damp Time ms
D-3-6521	1.66	207	593	1.0
D-3-6522	1.55	175	Bomb Failed to Ignite	
D-3-6523	1.62	18[illegible]	582	0.9
D-3-6524	1.62	224	587	1.0
D-3-6525	1.66	220	689	1.1
D-3-6526	1.72	18[illegible]	772	1.1
D-3-6527	1.70	201	708	1.2


10 ms
FVIP
1000 H3
OVIP
Acc Z
Acc Y
Acc X
TYPICAL BOMB TEST
TEST 6514 D-3
3-6-73
K Pc 4
K Pc 3
Pc (TABER)
BOMB SW PIP
FIRE SW PIP

SUMMARY OF TEST DATA

INJECTOR S/N FT-2A

ρ = 13.9%

Test No (B-1-)	O/F	P_c (psia)	He SAT	C* (FPS)	$I_{sp\infty}$ (sec)	$I_{sp\infty}$ (e=40) (sec)	T_{MAX}-Throat (°F)
702	1.50	201.3	YES	5308	289.8	293.3	
703	1.59	200.1	YES	5303	290.7	294.2	
704	1.73	200.0	YES	5293	291.0	294.5	
705	1.63	220.2	YES	5331	292.7	296.2	
706	1.64	178.5	YES	5315	290.7	294.2	
707	1.60	202.5	YES	5332	292.1	295.6	1975
708	1.57	177.9	NO	5304	291.7	295.2	
709	1.70	197.6	NO	5295	292.3	295.8	
710	1.61	198.5	NO	5307	292.3	295.8	
711	1.50	198.3	NO	5298	291.1	294.6	
712	1.58	216.4	NO	5346	294.8	298.3	
713	1.62	199.4	NO	5305	292.4	29[illegible].6	1963
714	1.62	200.9	NO	5304	291.9	295.4	
715	1.60	201.5	NO	5307	291.8	295.3	2010

NOTES: $I_{sp\infty}$ (e = 40) is projected from the measured values at e = 31.

All tests are 5-seconds duration except where T_{MAX} is indicated which is 30-seconds duration.

SUMMARY OF TEST DATA

INJECTOR S/N RDV-DD-PR-2B

$e = 31$

$\rho = 13.5\%$

TEST NO.	DATA PT.	O/F	P_c-PSIA	C*-FPS	$I_{sp\infty}$ SEC	$I_{sp\infty}$ (e=40) SEC	T_{MAX}(Throat) °F
B-1-637	-0.5	1.61	179.4	5271	289.8	293.3	
B-1-638	4.5	1.59	179.8	5287	290.7	294.2	1998
	29.8	1.59	179.1	5301	290.9	294.4	
B-1-639	-0.5	1.64	199.6	5271	290.6	294.1	
B-1-640	4.5	1.46	200.2	5287	290.3	293.8	1970
	29.8	1.46	199.9	5288	289.3	292.8	
B-1-641	4.5	1.60	198.7	5287	291.3	294.8	1959
	29.3	1.60	198.3	5279	289.8	293.3	

NOTES: $I_{sp\infty}$ (e = 40) is projected from the measured values at e = 31.

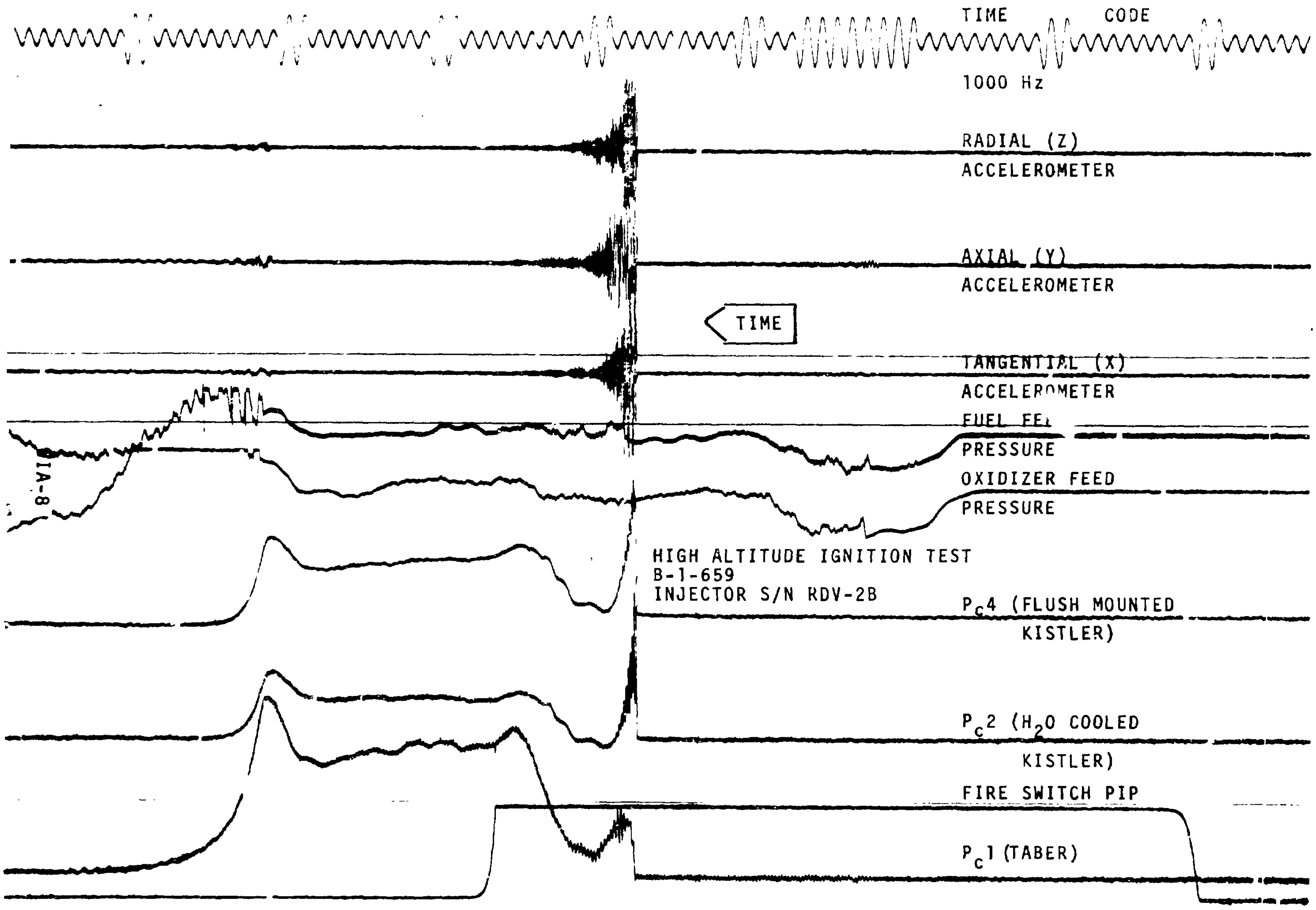
Tests with T_{max} specified were of 30-seconds duration. All others are 5 seconds duration.

SUMMARY OF HIGH ALTITUDE IGNITION TEST DATA

INJECTOR S/N RDV-DD-PR-2B

TEST NO	TEST CONDITIONS O/F	TEST CONDITIONS P_c-psia	TEST CONDITIONS He Sat Prop.	Alt. (K-Ft)	OFT-^oF	FFT-^oF	Inj. Temp.-^oF	P_c Spike psia
B-1-656	1.6	200	NO	269	39	41	40	706
657				262	39	41	37	573
658				262	38	40	36	662
659				266	38	40	34	720
660				277	38	40	34	662
661				277	37	39	33	647
662				277	37	39	33	711
663				275	37	39	32	706
664				275	37	39	36	637
665	1.6	200	NO	275	37	39	37	549
667	1.6	200	YES	289	36	41	44	458
668				283	36	41	42	434
669				284	36	40	41	491
670				288	36	41	40	614
671	1.6	200	YES	288	36	40	39	713
672	1.6	220	YES	287	36	40	38	670
673				294	36	39	37	590
674				294	35	39	37	604
675				291	35	38	36	717
676	1.6	220	YES	289	33	39	36	585
677	1.5	200	YES	288	35	38	36	694
678				279	35	38	36	656
679				279	36	38	36	684
680				278	38	39	36	684
681	1.5	200	YES	275	39	40	36	647

	TEST CONDITIONS							
TEST NO.	O/F	P_c-psia	He Sat Prop.	Alt. (K-Ft)	OFT-°F	FFT-°F	Inj. Temp.-°F	P_c Spike psia
B-1-682	1.7	200	YES	273	39	38	35	637
683				268	39	40	35	585
684				268	39	40	35	567
685				288	40	42	36	553
686	1.7	200	YES	273	39	41	36	659
687	1.6	180	YES	276	39	39	35	611
688				267	39	40	35	495
689				266	39	39	35	596
690				268	39	39	34	649
691	1.6	180	YES	269	39	39	34	630
692	1.6	200	YES	268	39	39	34	567
693				268	39	39	34	663
694				268	39	39	34	702
695				264	39	39	34	688
696				264	39	39	34	693
697				267	39	38	34	668
698				267	39	38	34	582
699				270	39	38	34	644
700				271	38	38	34	649
701	1.6	200	YES	271	38	38	34	678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FUEL FEED
PRESSURE
OXIDIZER FEED
PRESSURE
HIGH ALTITUDE IGNITION TEST
B-1-659
INJECTOR S/N RDV-2B
P_c4 (FLUSH MOUNTED
KISTLER)
P_c2 (H_2O COOLED
KISTLER)
FIRE SWITCH PIP
P_c1 (TABER)

PART B

ENGINE S/N FT-2B TEST DATA

TEST SUMMARY-ENGINE S/N FT-2A POST ENVIRONMENTAL CYCLES

$\epsilon = 40$

DATE	TEST NO.	I_T (LB-SEC) (50 MS PUL)	σ (LB-SEC)	O/F	P_c (PSIA)	C* FT/SEC	$I_{sp\infty}$ (SEC)	Remarks
6/20/73	B-1-780			1.60	198.7	5283	296.5	Baseline test prior to environmental testing.
6/21/73	B-1-783	21.3	0.7					" "
6/21/73	B-1-784	20.0	0.4					" "
7/11/73	B-1-809	22.8	0.3					Post Cycle #1.
7/11/73	B-1-810			1.64	199.2	5254	296.6	Post Cycle #1.
7/19/73	B-1-825	21.1	0.4					Post Cycle #2.
7/19/73	B-1-826			1.58	201.4	5261	296.0	Post Cycle #2.
7/25/73	B-1-827	23.6	0.3					Post Cycle #3.
7/25/73	B-1-828			1.64	199.2	5274	295.6	"
7/27/73	B-1-829	23.3	0.4					Post Cycle #4.
7/27/73	B-1-830			1.58	198.8	5288	295.6	"
7/30/73	B-1-831	24.2	0.3					Post Cycle #5.
7/30/73	B-1-832			1.64	196.9	5270	295.5	"

NOTE: $I_{sp\infty}$ and I_T are based on measured thrust.

DATE DATE	TEST NO.	I_T (LB-SEC)(50 MS PULSES EPW)	σ (LB-SEC)	O/F	P_c (PSIA)	C* FT/SEC	$I_{sp\infty}$ (SEC)	Remarks
7/31/73	B-1-833	23.5	0.3					Post Cycle #6.
7/31/73	B-1-834			1.60	198.1	5302	295.7	"
8/3/73	B-1-835	23.2	0.6					Post Cycle #7.
8/3/73	B-1-836			1.61	196.8	5279	295.2	"
8/8/73	B-1-839	23.8	0.3					Post Cycle #8.
8/8/73	B-1-840			1.61	198.5	5268	295.7	"
8/13/73	B-1-846	23.5	0.4					Post Cycle #9.
8/13/73	B-1-847			1.63	193.3	5272	294.1	"
8/15/73	B-1-849	23.8	0.3					Post Cycle #10.
8/15/73	B-1-850			1.58	193.9	5293	294.9	"

NOTE: $I_{sp\infty}$ and I_T are based on measured thrust.

PULSE PERFORMANCE CHARACTERIZATION SERIES
TEST SUMMARY - ENGINE S/N FT-2A
$\epsilon = 40$

Test No.	O/F	P_c	He SAT.	I_T (LB-SEC)	σ (LB-SEC)	PULSE $I_{SP\infty}$ (SEC)
B-1-783	1.6	200	No	21.27	.7	200.9
-784	1.6	200	No	19.85	.4	-
-851	1.6	200	No	21.09	.3	191.7
-852	1.5	200	No	21.10	.2	186.0
-908	1.6	200	No	26.95	.2	197.5
-909	1.6	200	No	27.21	.3	202.2
-910	-	-	No	-	-	MALFUNCTION
-911	-	-	No	-	-	MALFUNCTION
-912	1.6	220	No	29.44	.3	207.4
-913	1.6	220	No	29.41	.2	207.7
-914	1.6	180	No	26.43	.2	199.9
-915	1.6	180	No	26.79	.2	206.1
-916	-	-	Yes	BLEED IN TEST		
-917	1.6	220	Yes	26.73	.2	200.3
-918	1.6	220	Yes	26.54	.3	201.1
-919	1.6	180	Yes	25.80	.3	198.6
-920	1.6	180	Yes	25.85	.3	201.3
-921	1.6	200	Yes	26.24	.2	196.2
-922	-	-	Yes	-	-	MALFUNCTION
-923	-	-	Yes	-	-	MALFUNCTION
-924	LOST OX FLOW	200	Yes	29.74	.5	-

Note: $I_{SP\infty}$ and I_T are based on measured thrust.

Pulse Frequency for all tests is 1 cycle per second.

Electrical Pulse Width is 50 milliseconds for all tests.

WORST CASE MISSION DUTY CYCLE TEST SUMMARY

ENGINE S/N FT-2A

$\epsilon = 40$

Test No. B-1-925 Burn No.	Duration (Sec)	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SEC)	$T_{THROAT(MAX)}$ (^oF)
0-P	10-50 MS PULSES					
1		1.6	201	5300	295.8	1819
1-P	10-50 MS PULSES					
2		1.7	200	5299	295.7	1820
2-P	10-90 MS PULSES					
3		1.7	201	5299	295.7	1818
3-P	10-50 MS PULSES					
4		1.7	201	5299	295.7	1829
4-P	10-170 MS PULSES					
5		1.7	201	5298	295.7	1817
5-P	10-50 MS PULSES					
6		1.6	204	5311	296.4	1794
6-P	10-90 MS PULSES					
7		1.7	201	5297	295.6	1802
7-P	10-50 MS PULSES					
8		1.6	205	5314	296.6	1783
8-P	10-170 MS PULSES					
9		1.7	200	5296	295.6	1783
9-P	10-50 MS PULSES					
10		1.7	200	5295	295.5	1797
10-P	10-90 MS PULSES					

Note: $I_{SP\infty}$ is based on measured thrust.

BASELINE TESTS

ENGINE S/N FT-2A

$\epsilon = 40$

Test No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SEC)	$T_{THROAT(MAX)}$ (^oF)	I_T (LB-SEC)	(LB-SEC)
B-1-926	1.6	200				23.8	0.2
B-1-927	1.7	200	5294	295.5	1749		

Note: $I_{SP\infty}$ and I_T are based on measured thrust.

I_T is the average value for 25 pulses of 50 MS electrical pulse width.

PART C

ENGINE S/N RDV-2B

TEST DATA

PROFILE A

THERMAL DUTY CYCLE TEST SUMMARY

ENGINE S/N RDV-2B
e = 40

Test No. B-1-785-	O/F	P_c (psia)	C* (Ft/Sec)	$I_{sp\infty}$ (Sec)	Pulse EPW (ms)	Pulse Freq. (cps)	I_T (Lb-Sec)	σ (Lb-Sec)
A	1.57	195.9	5254	294.4				
B	1.57	195.8	5255	293.7				
C	1.57	196.1	5266	294.3				
D	1.56	196.4	5267	294.1				
E					50	5	QUESTIONABLE DATA	
F	1.56	196.5	5277	294.6				
G					50	2	24.1	0.3
H	1.56	196.8	5278	294.5				
I					50	1	24.4	0.3
J	1.56	197.0	5283	294.5				
K					50	5	26.0	1.3
L	1.56	196.8	5275	293.7				
M					50	2	23.9	0.3
N	1.56	197.5	5292	294.7				
O					50	1	24.3	0.3
P	1.56	197.5	5298	294.9				

NOTE: $I_{sp\infty}$ and I_T are based on measured thrust.

PROFILE A

PULSE PERFORMANCE CHARACTERIZATION SERIES

TEST SUMMARY

ENGINE S/N RDV-2B

e = 40

TEST NO.	Pulse Freq. (CPS)	Pulse EPW (MS)	I_T (LB-SEC)	σ (LB-SEC)	Pulse $I_{sp\infty}$ (SEC)
B-1-786	1	50	26.8	0.5	QUESTIONABLE FLOW DATA
B-1-787	1	50	26.7	0.2	200.1
B-1-788	1	50	26.8	0.2	200.9
B-1-789	5	50	31.6	0.8	253.5
B-1-790	5	50	31.6	0.7	253.7
B-1-791	0.2	50	24.6	0.4	198.4
B-1-792	1	91	50.9	0.3	238.4
B-1-793	1	92	50.7	0.3	
B-1-794	1	170	96.1	1.8	256.7
B-1-795	1	170	94.7	0.4	
B-1-796	1	36	10.3	4.8	126.1
B-1-797	1	36	9.9	4.7	
B-1-798	1	28	2.5	0.15	91.7
B-1-799	1	28	2.6	0.13	

NOTE: $I_{sp\infty}$ and I_T are based on measured thrust.

PROFILE A

WORST CASE MISSION DUTY CYCLE TEST SUMMARY

ENGINE S/N RDV-2B-1
e = 35

Test B-1-807 Burn No.	Duration (Sec)	O/F	P_c (psia)	C* (Ft/Sec)	$I_{sp\infty}$ (Sec)	$I_{sp\infty}$ (e = 40) (Sec)	$T_{Throat(Max)}$ ^oF
0-P	10-50 MS PULSES (1 CPS)						
1	103.4	1.61	196.9	5269	292.4	295.6	2159
1-P	10-50 MS PULSES (1 CPS)						
2	95.4	1.60	195.4	5244	291.5	294.7	2250
2-P	10-90 MS PULSES (1 CPS)						
3	97.8	1.59	197.1	5299	290.6	293.8	1937
3-P	10-50 MS PULSES (1 CPS)						
4	92.5	1.59	196.5	5283	289.9	293.1	1945
4-P	10-170 MS PULSES (1 CPS)						
5	99.8	1.60	196.7	5286	289.9	293.1	1951
5-P	10-50 MS PULSES (1 CPS)						
6	99.9	1.58	197.0	5281	292.7	295.9	1859
6-P	10-90 MS PULSES (1 CPS)						
7	100.3	1.61	194.3	5288	291.8	295.0	2235
7-P	10-50 MS PULSES (1 CPS)						
8	95.0	1.60	195.7	5281	293.4	296.6	1321
8-P	10-170 MS PULSES (1 CPS)						
9	94.9	1.60	192.3	5287	291.5	294.7	1909
9-P	10-50 MS PULSES (1 CPS)						
10	81.2	1.60	192.9	5303	291.5	294.7	1873

NOTE: $I_{sp\infty}$ is based on measured thrust.

PROFILE B

ENDURANCE TEST SUMMARY

ENGINE S/N RDV-2B-2

$\epsilon = 33$

DURATION OF TEST (B-1-848) - 600 SEC.

Data Point (Sec.)	O/F	P_c (PSIA)	C* (FT/SEC)	$I_{sp\infty}$ (SEC)	$I_{sp\infty}$ $\epsilon = 40$ (SEC)	T_{THROAT} (MAX) (^oF)
4.5	1.59	194.3	5290	290.7	294.1	1097
10	1.59	194.3	5285	290.8	294.2	1645
30	1.59	194.2	5283	291.2	294.6	2058
60	1.59	194.1	5281	291.2	294.6	2180
120	1.59	193.7	5275	291.4	294.8	2225
180	1.59	193.6	5278	291.5	294.9	2228
240	1.59	193.5	5282	291.6	295.0	2198
300	1.59	193.3	5282	291.5	294.9	2219
360	1.59	193.1	5283	291.5	294.9	2203
420	1.59	193.1	5287	291.6	295.0	2193
480	1.59	192.8	5286	291.6	295.0	2182
540	1.59	192.5	5284	291.4	294.8	2173
599.4	1.59	192.4	5287	291.5	294.9	2162

NOTE: $I_{sp\infty}$ is based on measured thrust.

PROFILE C

T ERMAL DUTY CYCLE TEST SUMMARY

ENGINE RDV-2B-1 - e = 35/1

Test No. B1-811-	O/F	P_c PSIA	C* (FPS)	$I_{sp\infty}$ (SEC)	$I_{sp\infty}$ e = 40 (SEC)	PULSE EPW (MS)	PULSE FREQ. (CPS)	I_T (LB-SEC)	σ (LB-SEC)
A	1.71	199.4	5223	291.3	294.6				
B	1.72	199.1	5207	291.3	294.6				
C	1.71	199.6	5228	291.5	294.8				
D	1.71	199.9	5239	291.5	294.8				
E						51	5	29.33	1.2
F	1.71	200.3	5247	291.1	294.4				
G						51	2	26.87	0.4
H	1.72	200.6	5251	291.7	295.0				
I						51	1	26.78	0.3
J	1.71	200.7	5258	291.2	294.5				
K						50	5	27.64	1.5
L	1.71	201.0	5276	291.6	294.9				
M						50	2	23.52	0.5
N	1.70	201.2	5289	292.0	295.3				
O						51	1	24.02	0.9
P	1.69	201.0	5291	292.1	295.4				

NOTE: $I_{sp\infty}$ and I_T are based on measured thrust.

PROFILE "C"

PULSE PERFORMANCE CHARACTERIZATION SERIES

TEST SUMMARY - ENGINE RDV-2B-1 - ϵ = 35/1

Test No.	Pulse Freq. (CPS)	Pulse EPW (MS)	I_T (LB-SEC)	σ (LB-SEC)	Pulse $I_{sp\infty}$ (SEC)	$I_{sp\infty}$ ϵ = 40 (SEC)
B-1-812	1	51	32.8	3.2	208.5	210.8
813	1	51	30.8	.4	211.1	213.5
814	5	51	30.0	4.0	221.4	223.9
815	0.2	51	29.4	.2	198.1	200.3
816	1	92	53.9	.3	233.7	236.3
817	1	173	100.3	.3	251.4	254.2
818	1	36	19.3	.2	177.6	179.6
819	1	28	11.1	1.6	139.1	140.7
820	5	51	30.0	.4	224.4	226.9
821	1	92	52.7	.2	255.7	228.2
822	1	172	99.3	3.4	244.5	247.3
823	1	36	20.0	.2	178.2	180.2
824	1	27	6.7	1.1	115.5	116.8

NOTE: $I_{sp\infty}$ and I_T are based on measured thrust.

PROFILE "C"

WORST CASE MISSION DUTY CYCLE TEST SUMMARY

ENGINE RDV-2B-2 - ϵ = 33/1

TEST NO. B-1-845 BURN NO.	DURATION (SEC)	O/F	P_c (PSIA)	C* (FT/SEC)	$I_{sp\infty}$ (SEC)	$I_{sp\infty}$ (ϵ=40/1) (SEC)	$T_{THROAT(MAX)}$ (°F)
0-P	10-50MS PULSES						
1	95.2	1.62	197.2	5275	291.3	294.7	1898
1-P	10-50MS PULSES						
2	65.2	1.63	198.7	5262	293.6	297.1	1940
2-P	10-90MS PULSES						
3	95.2	1.63	199.9	5270	291.2	294.6	1957
3-P	10-50MS PULSES						
4	90.[illegible]	1.61	202.0	5280	290.7	294.1	
4-P	10-170MS PULSES						
5	89.6	1.56	199.8	5299	291.6	295.0	1930
5-P	10-50MS PULSES						
6	89.7	1.61	199.5	5282	292.3	295.8	1976
6-P	10-90MS PULSES						
7	89.9	1.61	200.8	5283	292.9	296.4	2006
7-P	10-50MS PULSES						
8	90.2	1.57	204.7	5304	291.0	294.4	1963
8-P	10-170 MS PULSES						
9	90.2	1.61	203.8	5295	293.2	296.7	2005
9-P	10-50 MS PULSES						
10	89.7	1.61	205.4	5298	292.8	296.3	2024
10-P	10-90 MS PULSES						

NOTE: $I_{sp\infty}$ and I_T are based on measured thrust.

TABLE 3-1. PROFILE "D" - PULSE PERFORMANCE CHARACTERIZATION SERIES

TEST SUMMARY - ENGINE S/N RDV-2B-2

ϵ = 33, PULSE EPW = 50 MS

TEST NO.	P_C (PSIA)	O/F	PULSE FREQ. (CPS)	I_T (LB-SEC)	σ (LB-SEC)	PULSE $I_{SP\infty}$ (SEC)	$I_{SP\infty}$ ϵ = 40 (SEC)	He SAT.	$T_{PROPS.}$ (°F)
B-1-853	192.3	1.578	4	SEC CHECKOUT TEST					
854	211.4	1.565	1	28.9	.3	200.2	202.6	No	75
855	173.0	1.571	1	27.1	2.0	196.6	199.0		75
856	196.0	1.627	1	28.0	.2	195.6	197.9		75
857	189.7	1.526	1	28.5	.6	179.9	182.1		40
858	173.1	1.419	1	28.3	2.4	190.5	192.8		40
859	215.6	1.676	1	27.9	.2	188.7	191.0		40
860	191.8	1.583	1	29.2	.3	196.1	198.5		40
861	191.6	1.564	1	27.6	2.8	203.3	205.7		110
862	214.8	1.456	1	27.8	2.8	205.8	208.3		110
863	169.8	1.604	1	PV MALFUNCTION					110
864			1	27.9	.2	184.1	186.3	No	110
865	192.2	1.559	4	SEC CHECKOUT TEST				Yes	110
866	192.7	1.580	1	29.3	.3	216.7	219.3		110
867	215.5	1.447	1	29.7	.2	-	-		110
868	172.7	1.612	1	QUESTIONABLE DATA					110
869	196.4	1.614	1	29.1	.2	185.0	187.2		110
870	216.0	1.627	1	29.8	.3	216.7	219.3		75
871	173.4	1.570	1	23.3	2.3	185.4	187.6		75
872	194.2	1.575	1	29.1	.2	202.0	204.4		75
873	195.0	1.558	1	29.2	.2	202.3	204.7		75
874	196.2	1.591	1	30.0	.3	192.2	194.5		40
875	176.3	1.488	1	29.8	.2	192.0	194.3		40
876	216.8	1.694	1	28.9	.2	176.0	178.1		40
877	214.3	1.697	1	27.5	3.6	215.0	217.6		40
878			1	26.2	1.3	-	-	Yes	40

NOTE: I_T and $I_{SP\infty}$ are based on measured thrust.

PROFILE "D" - STEADY STATE PERFORMANCE CHARACTERIZATION SERIES

TEST SUMMARY - ENGINE RDV-2B-2 - ϵ = 33

TEST NO.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ ϵ = 40 (SECS)	He SAT.	T_{PROPS} (^oF)
B-1-879	1.55	196	5244	289.0	292.5	No	40
880	1.63	200	5224	288.5	292.0		
881	1.71	201	5200	287.3	290.7		
882	1.71	201	5198	287.4	290.8		
883	1.51	203	5262	289.6	293.1		
884	1.50	202	5260	289.4	292.9		
885	1.63	214	5241	289.6	293.1		
886	1.61	214	5250	290.1	293.6		
887	1.63	191	5227	288.2	291.7		
888	1.69	189	5194	286.8	290.2	No	
889	1.54	202	5266	290.4	293.9	Yes	
890	1.61	200	5243	289.6	293.1	Yes	
891	1.60	202	5245	289.6	293.1	Yes	40
892	1.60	197	5274	291.0	294.5	No	75
893	1.53	197	5350	294.9	298.4		75
894	1.62	196	5296	292.7	296.2		75
895	1.59	194	5317	293.7	297.2		110
896	1.58	194	5293	292.4	295.9		
897	1.47	196	5313	292.7	296.2		
898	1.47	194	5302	291.9	295.4		
899	1.69	193	5271	291.8	295.3		
900	1.66	193	5279	292.1	295.6		
901	1.58	174	5290	291.2	294.7		
902	1.59	175	5289	291.2	294.7		
903	1.56	DATE NOT RECORDED					
904	1.56	214	5305	292.5	296.0	No	
905	1.56	216	5301	292.6	296.1	Yes	
906	1.59	197	5307	292.7	296.2	Yes	
907	-	LOST DATA		-	-	Yes	110

Note: $I_{SP\infty}$ is based on measured thrust.

PROFILE "E" - WORST CASE MISSION DUTY CYCLE
TEST SUMMARY AT MAXIMUM CONDITIONS

ENGINE S/N RDV-2B-2; ϵ = 33

TEST NO. B-1-929 BURN NO.	DURATION (SEC)	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SEC)	$I_{SP\infty}$ ϵ = 40 (SEC)	T_{THROAT} (MAX) (°F)
0-P	10-50 MS PULSES (1 CPS)						
1	100.2	1.65	208.4	5290	292.4	295.9	2181
1-P	10-50 MS PULSES (1 CPS)						
2	95.1	1.65	209.5	5251	295.1	298.6	2301
2-P	10-95 MS PULSES (1 CPS)						
3	95.2	1.64	211.1	5248	294.7	298.2	2312
3-P	10-50 MS PULSES (1 CPS)						
4	95.3	1.63	212.9	5244	294.4	297.9	2301
4-P	10-170 MS PULSES (1 CPS)						

Note: $I_{SP\infty}$ is based on measured thrust.

Propellants are helium saturated and at 110-116°F.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 1 TEST SUMMARY
ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-930- Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
1	1.61	193	5277	292.5	296.0
2	1.60	195	5282	292.0	295.5
3	1.60	196	5285	292.7	296.2
4	1.60	196	5287	292.6	296.1
5	1.60	196	5284	292.2	295.7
5-P	10-50 MS PULSES (1 CPS)				
6	1.6	195	5287	291.8	295.3
7	1.6	195	5284	292.6	296.1
8	1.6	195	5283	292.4	295.9
9	1.6	195	5286	292.2	295.7
10	1.6	195	5285	292.3	295.8
10-P	10-50 MS PULSES (1 CPS)				
11	1.6	195	5290	292.1	295.6
12	1.6	195	5286	292.4	295.9
13	1.6	195	5289	292.4	295.9
14	1.6	195	5288	292.1	295.6
15	1.6	195	5280	292.1	292.1
15-P	10-50 MS PULSES (1 CPS)				
16	1.6	196	5290	291.9	295.4
17	1.6	195	5288	292.3	295.8
18	1.6	196	5292	292.6	296.1
19	1.6	196	5298	292.8	296.3
20	1.6	196	5295	292.7	296.2
20-P	10-50 MS PULSES (1 CPS)				
21	1.5	197	5298	292.2	295.7
22	1.6	197	5295	292.5	296.0
23	1.6	197	5296	292.7	296.2

NOTE: $I_{SP\infty}$ is based on measured thrust.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 2
TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-931- Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
1	1.5	197	5284	292.0	295.5
2	1.6	194	5268	291.9	295.4
3	1.6	195	5277	291.9	295.4
4	1.6	195	5273	291.9	295.4
5	1.6	195	5273	291.8	295.3
5-P	10-50 MS PULSES (.2 CPS)				
6	1.6	196	5283	292.2	295.7
7	1.6	195	5274	292.0	295.5
8	1.6	195	5273	292.2	295.7
9	1.6	195	5273	291.8	295.3
10	1.6	195	5273	291.8	295.3
10-P	10-50 MS PULSES (.2 CPS)				
11	1.6	197	5286	293.8	297.3
12	1.6	195	5277	292.4	295.9
13	1.6	195	5283	292.3	295.8
14	1.6	195	5284	292.3	295.8
15	1.6	195	5284	292.4	295.9
15-P	10-50 MS PULSES (.2 CPS)				
16	1.5	196	5290	292.5	296.0
17	1.6	196	5284	292.5	296.0
18	1.6	196	5290	292.7	296.2
19	1.6	196	5287	292.5	296.0
20	1.5	196	5289	292.0	295.5
20-P	10-50 MS PULSES (.2 CPS)				
21	1.5	197	5293	292.1	295.6
22	1.6	196	5293	292.0	295.5
23	1.6	196	5295	292.2	295.7
24	1.6	196	5295	292.4	295.9
25	1.6	196	5294	292.5	296.0
25-P	10-50 MS PULSES (.2 CPS)				
26	1.6	198	5297	-	-

NOTE: $I_{SP\infty}$ is based on measured thrust.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 3
TEST ENGINE SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-932- Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	197	5280	291.9	295.4
7	1.6	197	5288	292.8	296.3
8	1.6	197	5284	292.2	295.7
9	1.6	197	5282	292.1	295.6
10	1.6	197	5283	292.1	295.6
10-P	10-50 MS PULSES (5 CPS)				
11	1.6	197	5287	292.3	295.8
12	1.6	197	5282	292.0	295.5
13	1.6	197	5287	292.3	295.8
14	1.6	197	5282	292.0	295.5
15	1.6	197	5287	292.2	295.7
15-P	10-50 MS PULSES (5 CPS)				
16	1.6	197	5285	292.3	295.8
17	1.6	197	5287	292.2	295.7
18	1.6	198	5289	292.2	295.7
19	1.6	198	5291	292.4	295.9
20	1.6	198	5288	292.2	295.7
20-P	10-50 MS PULSES (5 CPS)				
21	1.5	199	5292	292.3	295.8
22	1.6	199	5292	292.3	295.8
23	1.6	199	5295	292.4	295.9
24	1.6	199	5295	292.4	295.9
25	1.6	199	5294	292.3	295.3
25-P	10-50 MS PULSES (5 CPS)				
26	1.5	199	5297	292.5	296.0
27	1.5	198	5294	292.7	296.2
28	1.5	199	5296	292.4	295.9
29	1.5	198	5297	292.4	295.9
30	1.5	200	5299	292.6	296.1
30-P	10-50 MS PULSES (5 CPS)				
1	1.5	200	5298	292.3	295.8
2	1.6	199	5294	292.4	295.9
3	1.6	199	5294	292.3	295.8

Note: $I_{SP\infty}$ is based on measured thrust.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 4
TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-933- Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.5	197	5289	291.6	295.1
7	1.4	192	5281	290.8	294.3
8	1.4	191	5275	290.3	293.8
9	-	-	-	-	-
10	1.4	192	5277	290.6	294.1
10-P	10-90 MS PULSES (1 CPS)				
11	1.5	198	5293	292.1	295.6
12	1.5	199	5286	292.2	295.7
13	1.5	199	5286	292.3	295.8
14	1.5	199	5285	292.2	295.7
15	1.5	199	5286	292.3	295.8
15-P	10-90 MS PULSES (1 CPS)				
16	1.5	200	5289	292.3	295.8
17	1.5	199	5288	289.2	292.7
18	1.5	199	5287	289.1	292.6
19	1.5	199	5291	292.4	295.9
20	1.5	199	5288	292.3	295.8
20-P	10-90 MS PULSES (1 CPS)				
21	1.5.	200	5294	292.4	295.9
22	1.5	200	5293	292.5	296.0
23	1.5	200	5292	292.4	295.9
24	1.5	200	5291	292.4	295.9
25	1.5	200	5292	292.5	296.0
25-P	10-90 MS PULSES (1 CPS)				
26	1.5	201	5295	292.6	296.1
27	1.5	200	5293	292.6	296.1
28	1.5	200	5294	292.6	296.1
29	1.5	201	5295	292.7	296.2
30	1.5	201	5299	292.8	296.3
30-P	10-90 MS PULSES (1 CPS)				
1	1.5	202	5295	292.5	296.0
2	1.6	201	5292	293.0	296.5
3	1.6	201	5293	293.2	296.7
4	1.6	201	5291	293.2	296.7
5	1.6	201	5290	293.1	296.6
5-P	10-90 MS PULSES (1 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 5
TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

TEST NO. B-[illegible]-934 START NO.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	189	5199	294.1	297.6
7	1.6	189	5189	294.5	298.0
8	1.6	189	5191	294.0	297.5
9	1.6	191	5273	293.5	297.0
10	1.6	195	5278	293.8	297.3
10-P	10-90 MS PULSES (0.2 CPS)				
11	1.6	195	5279	292.0	295.5
12	1.6	195	5278	292.1	295.6
13	1.6	194	5277	291.8	295.3
14	1.6	194	5276	291.4	294.9
15	1.6	195	5280	291.6	295.1
15-P	10-90 MS PULSES (0.2 CPS)				
16	1.5	195	5281	291.6	295.1
17	1.5	195	5285	291.6	295.1
18	1.5	195	5282	291.6	295.1
19	1.5	196	5284	291.6	295.1
20	1.5	196	5287	291.7	295.2
20-P	10-90 MS PULSES (0.2 CPS)				
21	1.5	196	5289	291.6	295.1
22	1.5	196	5290	291.7	295.2
23	1.5	197	5292	291.6	295.1
24	1.5	197	5292	291.8	295.3
25	1.5	197	5294	291.5	295.0
25-P	10-90 MS PULSES (0.2 CPS)				
26	1.5	197	5296	291.5	295.0
27	1.5	197	5294	291.5	295.0
28	1.5	197	5295	291.4	294.9
29	1.5	198	5294	291.4	294.9
30	-	-	-	-	-
30-P	10-90 MS PULSES (0.2 CPS)				
1	1.5	198	5294	291.4	294.9
2	1.5	197	5292	292.3	295.8
3	1.5	197	5290	292.4	295.9
4	1.5	197	5288	292.4	295.9
5	-	-	-	-	-
5-P	10-90 MS PULSES (0.2 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 6
TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-935- Start No.	O/F	P_C (PSIA)	C^* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.5	197	5275	289.4	292.9
7		*			
8	1.5	196	5285	291.4	294.9
9		*			
10	1.5	196	5282	291.6	295.1
10-P	10-90 MS PULSES (5 CPS)				
11	1.5	197	5283	291.4	294.9
12		*			
13	1.5	196	5282	291.6	295.1
14		*			
15	1.5	196	5283	291.7	295.2
15-P	10-90 MS PULSES (5 CPS)				
16	1.5	197	5283	291.5	295.0
17		*			
18	1.5	197	5288	291.8	295.3
19		*			
20	1.5	197	5287	291.8	295.3
20-P	10-90 MS PULSES (5 CPS)				
21	1.5	198	5290	291.9	295.4
22		*			
23	1.5	197	5293	292.1	295.6
24		*			
25	1.5	197	5254	292.0	295.5
25-P	10-90 MS PULSES (5 CPS)				
26	1.5	198	5293	292.2	295.7
27		*			
28	1.5	198	5297	292.1	295.6
29		*			
30	1.5	198	5299	292.2	295.7
30-P	10-90 MS PULSES (5 CPS)				
1	1.5	198	5296	292.1	295.6
2		*			
3	1.5	196	5295	293.3	295.8
4		*			
5	1.5	196	5297	293.1	296.6
5-P	10-90 MS PULSES (5 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 7
TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-936- Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	196	5282	292.5	296.0
7	1.6	196	5275	292.9	296.4
8	1.7	194	5250	291.7	295.2
9	1.6	196	5277	292.0	295.5
10	1.6	195	5281	292.7	296.2
10-P	10-130 MS PULSES (1 CPS)				
11	1.5	193	5290	292.2	295.7
12	1.5	193	5289	292.0	295.5
13	1.6	193	5287	292.0	295.5
14	1.6	192	5287	291.9	295.4
15	1.6	193	5289	292.0	295.5
15-P	10-130 MS PULSES (1 CPS)				
16	1.5	193	5293	292.3	295.8
17	1.5	193	5292	292.6	296.1
18	1.5	193	5296	292.1	295.6
19	1.5	193	5299	291.9	295.4
20	1.5	193	5297	291.6	295.1
20-P	10-130 MS PULSES (1 CPS)				
21	1.5	194	5298	291.6	295.1
22	1.5	194	5299	291.8	295.3
23	1.5	194	5299	291.7	295.2
24	1.5	194	5300	291.7	295.2
25	1.5	194	5298	291.4	294.9
25-P	10-130 MS PULSES (1 CPS)				
26	1.5	195	5300	291.6	295.1
27	1.5	195	5300	292.0	295.5
28	1.5	195	5305	292.1	295.6
29	1.5	195	5303	292.0	295.5
30	1.5	195	5303	292.2	295.7

Note: $I_{SP\infty}$ is based on measured thrust.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 8
TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-937 Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	198	5291	294.0	297.5
7		*			
8	1.8	192	5223	290.4	293.9
9		*			
10	1.6	198	5290	292.9	296.4
10-P	10-130 MS PULSES (0.2 CPS)				
11	1.6	202	5289	293.2	296.7
12		*			
13	1.6	202	5276	293.2	296.7
14		*			
15	1.6	202	5276	293.1	296.6
15-P	10-130 MS PULSES (0.2 CPS)				
16	1.6	204	5285	293.5	297.0
17		*			
18	1.6	203	5280	293.0	296.5
19		*			
20	1.6	203	5282	293.0	296.5
20-P	10-130 MS PULSES (0.2 CPS)				
21	1.6	205	5291	293.3	296.8
22		*			
23	1.6	203	5281	292.5	296.0
24		*			
25	1.6	203	5285	292.4	295.9
25-P	10-130 MS PULSES (0.2 CPS)				
26	1.6	205	5292	292.0	295.5
27		*			
28	1.6	203	5284	-	-
29		*			
30	1.7	202	5280	292.2	295.7
30-P	10-130 MS PULSES (0.2 CPS)				
1	1.6	203	5281	289.1	292.6
2		*			
3	1.7	201	5275	292.8	296.3
4		*			
5	1.7	201	5274	292.7	296.2
5-P					

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 9
TEST SUMMARY - ENGINE S/N RDV-2B-2

Test No. B-1-938 Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ $\epsilon = 33$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	199	5291	295.2	298.7
7		*			
8	1.6	199	5291	293.5	297.0
9		*			
10	1.6	199	5289	293.5	297.0
10-P	10-130 MS PULSES (5 CPS)				
11	1.6	199	5289	292.4	295.9
12		*			
13	1.6	198	5289	292.6	296.1
14		*			
15	1.6	198	5288	292.5	296.0
15-P	10-130 MS PULSES (5 CPS)				
16	1.6	198	5288	292.2	295.7
17		*			
18	1.6	197	5286	292.0	295.5
19		*			
20	1.6	197	5285	292.1	295.6
20-P	10-130 MS PULSES (5 CPS)				
21	1.6	198	5288	292.5	296.0
22		*			
23	1.6	198	5293	292.6	296.1
24		*			
25	1.6	198	5296	292.6	296.1
25-P	10-130 MS PULSES (5 CPS)				
26	1.6	198	5294	292.3	295.8
27		*			
28	1.6	198	5299	291.9	295.4
29		*			
30	1.6	199	5298	292.0	295.5
30-P	10-130 MS PULSES (5 CPS)				
1	1.6	198	5297	292.2	295.7
2		*			
3	1.5	198	5298	293.2	296.7
4		*			
5	1.5	199	5298	293.7	297.2
5-P	10-130 MS PULSES (5 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 10
TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-939- Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	197	5297	294.1	297.6
7		*			
8	1.6	197	5289	292.7	296.2
9		*			
10	1.6	196	5288	293.1	296.6
10-P	10-170 MS PULSES (1 CPS)				
11	1.6	197	5285	292.3	295.8
12		*			
13	1.6	197	5284	292.3	295.8
14		*			
15	1.6	197	5286	292.5	296.0
15-P	10-170				
16	1.6	197	5288	292.5	296.0
17		*			
18	1.6	197	5292	292.8	296.3
19		*			
20	1.5	198	5292	292.7	296.2
20-P	10-170 MS PULSES (1 CPS)				
21	1.5	198	5292	292.3	295.8
22		*			
23	1.6	199	5293	292.5	296.0
24		*			
25	1.6	199	5295	292.7	296.2
25-P	10-170 MS PULSES (1 CPS)				
26	1.6	199	5293	292.4	295.9
27		*			
28	1.6	199	5294	292.2	295.7
29		*			
30	1.6	199	5298	292.3	295.8
30-P	10-170 MS PULSES (1 CPS)				
1	1.6	199	5297	291.9	295.4
2		*			
3	1.6	199	5294	293.5	297.0
4		*			
5	1.6	199	5293	293.3	296.8
5-P	10-170 MS PULSES (1 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 11
TEST SUMMARY - ENGINE S/N EDV-2B-2
$\epsilon = 33$

Test No. B-1-940- Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	199	5294	294.4	297.9
7		*			
8	1.6	199	5290	293.5	297.0
9		*			
10	1.6	199	5285	293.6	297.1
10-P	10-170 MS PULSES (.2 CPS)				
11	1.6	199	5288	292.8	296.3
12		*			
13	1.6	199	5285	29[illegible].4	296.9
14		*			
15	1.6	199	5285	293.1	296.6
15-P	10-170 MS PULSES (.2 CPS)				
16	1.6	199	5286	292.9	296.4
17		*			
18	1.6	199	5288	293.[illegible]	296.8
19		*			
20	1.6	198	5290	293.[illegible]	296.9
20-P	10-170 MS PULSES (.2 CPS)				
21	1.6	198	5293	293.4	296.9
22		*			
23	1.6	199	5287	292.8	296.3
24		*			
25	1.6	199	5292	292.6	296.1
25-P	10-170 MS PULSES (.2 CPS)				
26	1.6	199	5296	292.9	296.4
27		*			
28	1.6	199	5293	292.6	296.1
29		*			
30	1.6	199	5298	292.8	296.3
30-P	10-170 MS PULSES (.2 CPS)				
1	1.6	199	5292	292.1	295.6
2		*			
3	1.6	199	5293	294.1	297.6
4		*			

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PROFILE "F" - THERMAL MISSION DUTY CYCLE
NO. 12 - TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-941 Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.4	192	5276	290.9	294.4
7		*			
8	1.4	193	5284	290.4	293.9
9		*			
10	1.4	193	5282	290.0	293.5
10-P	10-170 MS PULSES (5 CPS)				
11	1.5	198	5294	292.8	296.3
12		*			
13	1.5	198	5296	292.7	296.2
14		*			
15	1.5	198	5298	293.0	296.5
15-P	10-170 MS PULSES (5 CPS)				
16	1.5	198	5295	293.3	296.8
17		*			
18	1.5	199	5295	293.6	297.1
19		*			
20	1.5	198	5298	293.7	297.2
20-P	10-170 MS PULSES (5 CPS)				
21	1.5	199	5301	293.9	297.4
22		*			
23	1.5	198	5301	293.9	297.4
24		*			
25	1.5	198	5303	293.9	297.4

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 13
TEST SUMMARY - ENGINE S/N RDV-2B-2

Test No. B-1-942 Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS) $\epsilon = 33$	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	202	5297	293.9	297.4
7		*			
8	1.6	202	5294	294.0	297.5
9		*			
10	1.6	201	5285	293.2	296.7
10-P	10-50 MS PULSES (1 CPS)				
11	1.6	201	5291	293.4	296.9
12		*			
13	1.6	201	5293	293.3	296.8
14		*			
15	1.6	202	5284	293.8	297.3
15-P	10-50 MS PULSES (1 CPS)				
16	1.6	201	5288	293.5	297.0
17		*			
18	1.6	201	5292	292.8	296.3
19		*			
20	1.6	202	5293	292.8	296.3
20-P	10-50 MS PULSES (1 CPS)				
21	1.6	202	5294	293.5	297.0
22		*			
23	1.6	202	5300	293.1	296.6
24		*			
25	1.6	202	5304	293.3	296.8
25-P	10-50 MS PULSES (1 CPS)				
26	1.6	203	5304	293.6	297.1
27		*			
28	1.6	203	5307	293.2	296.7
29		*			
30	1.6	203	5310	293.0	296.5
30-P	10-50 MS PULSES (1 CPS)				
1	1.6	203	5311	293.7	297.2
2		*			
3	1.6	203	5307	293.3	296.8
4		*			
5	1.6	203	5307	293.4	296.9
5-P	10-50 MS PULSES (1 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 14
TEST SUMMARY - ENGINE S/N EDV-2B-2

Test No. B-1-943 Start No.	O/F	$\epsilon = 33$ P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	202	5278	293.9	297.4
7		*			
8	1.4	197	5283	292.2	295.7
9		*			
10	1.5	199	5277	292.2	295.7
10-P	10-50 MS PULSES (0.2 CPS)				
11	1.6	200	5275	293.1	296.6
12		*			
13	1.6	200	5268	292.7	296.2
14		*			
15	1.6	200	5269	293.3	296.8
15-P	10-50 MS PULSES (0.2 CPS)				
16	1.6	201	5271	293.2	296.7
17		*			
18	1.6	200	5270	292.6	296.1
19		*			
20	1.6	201	5273	293.0	296.5
20-P	10-50 MS PULSES (0.2 CPS)				
21	1.6	201	5280	293.2	296.7
22		*			
23	1.6	201	5281	292.8	296.3
24		*			
25	1.6	201	5281	291.5	295.0
25-P	10-50 MS PULSES (0.2 CPS)				
26	1.6	202	5284	293.3	296.8
27		*			
28	1.6	202	5288	292.8	296.3
29		*			
30	1.6	202	5287	292.9	296.4
30-P	10-50 MS PULSES (0.2 CPS)				
1	1.6	202	5291	293.5	297.0
2		*			
3	1.6	201	5287	293.6	297.1
4		*			
5	1.6	201	5288	293.4	296.9
5-P	10-50 MS PULSES (0.2 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 15
TEST SUMMARY - ENGINE S/N RDV-2B-2
$\epsilon = 33$

Test No. B-1-944 Start No.	O/F	P_c (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
6	1.6	200	5284	292.4	295.9
7		*			
8	1.6	199	5283	292.5	296.0
9		*			
10	1.6	199	5274	292.1	295.6
10-P	10-50 MS PULSES (5 CPS)				
11	1.6	198	5274	292.0	295.5
12		*			
13	1.6	198	5274	291.6	295.1
14		*			
15	1.6	198	5268	291.6	295.1
15-P	10-50 MS PULSES (5 CPS)				
16	1.6	198	5274	292.2	295.7
17		*			
18	1.6	198	5275	291.6	295.1
19		*			
20	1.6	199	5260	291.1	294.6
20-P	10-50 MS PULSES (5 CPS)				
21	1.6	200	5274	292.1	295.6
22		*			
23	1.6	202	5290	292.1	295.6
24		*			
25	1.6	203	5292	291.9	295.4
25-P	10-50 MS PULSES (5 CPS)				
26	1.6	203	5295	292.4	295.9
27		*			
28	1.6	203	5295	292.0	295.5
29		*			
30	1.6	203	5297	292.1	295.6
30-P	10-50 MS PULSES (5 CPS)				
1	1.6	203	5301	293.1	296.6
2		*			
3	1.6	203	5298	292.7	296.2
4		*			
5	1.6	203	5297	292.9	296.4
5-P	10-50 MS PULSES (5 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PROFILE "F" - THERMAL MISSION DUTY CYCLE NO. 16
TEST SUMMARY - ENGINE S/N RDV-2B-2

Test No. B-1-945 Start No.	O/F	P_C (PSIA)	C* (FPS)	$I_{SP\infty}$ (SECS)	$I_{SP\infty}$ $\epsilon = 40$ (SECS)
		$\epsilon = 33$			
6	1.6	201	5285	293.5	297.0
7		*			
8	1.6	199	5274	292.8	296.3
9		*			
10	1.6	199	5270	292.5	296.0
10-P	10-50 MS PULSES (5 CPS)				
11	1.6	200	5269	292.0	295.5
12		*			
13	1.6	200	5269	291.9	295.4
14		*			
15	1.6	200	5270	292.4	295.9
15-P	10-50 MS PULSES (5 CPS)				
16	1.6	200	5269	292.3	295.8
17		*			
18	1.6	200	5271	291.9	295.4
19		*			
20	1.6	200	5271	291.9	295.4
20-P	10-50 MS PULSES (5 CPS)				
21	1.6	201	5275	291.6	295.1
22		*			
23	1.6	200	5270	291.5	295.0
24		*			
25	1.6	200	5271	291.6	295.1
25-P	10-50 MS PULSES (5 CPS)				
26	1.6	201	5273	292.3	295.8
27		*			
28	1.6	201	5284	292.[illegible]	295.5
29		*			
30	1.6	201	5281	292.1	295.6
30-P	10-50 MS PULSES (5 CPS)				
1	1.6	201	5285	293.1	296.6
2		*			
3	1.6	200	5279	292.3	295.8
4		*			
5	1.6	200	5279	292.7	296.2
5-P	10-50 MS PJLSES (5 CPS)				

Note: $I_{SP\infty}$ is based on measured thrust.

*Data not reduced to minimize cost.

PART D

OFF DESIGN TESTS

HELIUM BUBBLE INGESTION TESTS - ENGINE S/N RDV-2B-2

$\epsilon = 33$

Test No.	O/F	Pc (psia)	C* (fps)	$I_{sp\infty}$ (sec)	$I_{sp\infty}$ $\mathcal{E} = 40$ (sec)	Remarks
B-1-953	1.51	195.2	5297	293.2	296.7	Baseline
B-1-954	1.53	195.0	5296	292.9	296.9	Oxidizer side helium bubble
	*1.44	192.3	5312	293.4	296.9	Ingestion test
B-1-955	1.52	194.9	5292	293.4	296.9	Fuel side helium bubble ingestion
	*1.65	191.9	5299	293.8	297.3	Test
B-1-956	1.52	195.0	5298	293.3	296.8	Helium bubble ingestion test
	*1.55	189.3	5319	294.2	297.7	Oxidizer and fuel sides simultaneously
B-1-957	1.53	196.3	5299	292.3	295.8	Oxidizer side helium bubble
	*1.45	192.6	5315	293.0	296.5	Ingestion Test
	1.53	195.2	5296	293.7	297.2	$T_{max} = 1997^{\circ}F$.
B-1-958	1.55	194.8	5297	291.7	295.2	Helium bubble ingestion test-Oxidizer
	*1.58	188.4	5313	293.8	297.3	and fuel sides simultaneously
	1.54	194.3	5290	292.6	296.1	$T_{max} = 2050^{\circ}F$.
B-1-959	1.54	195.2	5297	292.4	295.8	Fuel side helium bubble ingestion
	*1.66	191.2	5294	294.1	297.6	test
	1.54	194.2	5288	293.2	296.7	$T_{max} = 2084^{\circ}F$.

Note: $I_{sp\omega}$ is based on measured thrust. Tests B-1-957 through B-1-959 were 60-second duration tests.

* Performance during helium bubble flow.

LOW CHAMBER PRESSURE TESTS - ENGINE S/N RDV-2B-2

$\epsilon = 33$

Test No.	O/F	P_c (PSIA)	C* (FPS)	$I_{sp\infty}$ (SEC)	$I_{sp\infty}$ $\mathcal{E} = 40$ (SEC)	T_{max} (^oF)
B-1-960	1.57	197.5	5291	292.9	296.4	1988
B-1-961	1.58	179.6	5312	292.2	295.7	1988
B-1-962	1.61	147.8	5245	288.9	292.4	1908
B-1-963	1.48	114.8	4986	266.5	269.7	1505

Note: $I_{sp\infty}$ is based on measured thrust

OFF MIXTURE RATIO TESTS - ENGINE S/N RDV-2B-2

$\varepsilon = 33$

Test No.	O/F	P_C (PSIA)	C* (FPS)	$I_{sp\infty}$ (SEC)	$I_{sp\infty}$ $\varepsilon = 40$ (SEC)	T_{max} (^oF)
B-1-964	0.97	189.5	4937	270.9	274.2	
B-1-965	2.51	175.4	4660	259.4	262.5	
B-1-966	2.15	188.5	4939	275.7	279.0	
B-1-967	0.97	190.7	4963	271.7	275.0	937
B-1-968	2.16	188.5	4957	275.8	279.1	2287

Note: $I_{sp\infty}$ is based on measured thrust.